D180-18553-1

FINAL REPORT
January 1975

Concept Definition and System Analysis Study for a Solar Electric Propulsion Stage

Volume I

Executive Summary

Boeing Aerospace Company
Seattle Washington

(NASA-CR-161599) CONCEPT DEFINITION AND SYSTEM ANALYSIS STUDY FOR A SOLAR ELECTRIC PROPULSION STAGE. VOLUME 1: EXECUTIVE SUMMARY Final Report (Boeing Aerospace Co., Seattle, Wash.) 70 p
N81-70336
Unclas
00/20 39276

MSFC

Contract No. NAS 8-30921

DR No. MA-04

DPD No. 436

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FOREWORD

Electric propulsion has been recognized as an efficient method for space propulsion since the early 1960's. Related component development was started in that era and is continuing today. The basic components for an electric propulsion stage have proven successful in space applications.

Present studies for a Solar Electric Propulsion Stage (SEPS) were initiated in 1971 and have progressed to the present Phase A status. Technology related to development of a stage has advanced significantly, allowing design of a relatively high-performance vehicle. Study results indicate that SEPS has a logical place in space propulsion, both for planetary and Earth-orbital applications.

The "Concept Definition and Systems Analysis Study for a Solar Electric Propulsion Stage" was performed for the Marshall Space Flight Center by Boeing Aerospace Company. It was accomplished under NASA Contract NAS 8-30921. The information developed during the contract period is contained in this six-volume final report. The title of each volume is listed below.

Volume I	Executive Summary
Volume II	Mission and System Analysis
Volume III	Configuration and Subsystem Design and Analysis
Volume IV	Program Planning Data
Volume V	Cost Data
Volume VI	Systems Requirements Data Book

Volume I contains the executive summary of Boeing Phase A study effort. An overview of the SEPS program is included in the introduction. Other sections of this volume relate to study objectives, approach, limitations, and utilization of advanced space technology. The sections most pertinent to SEPS continued development are those dealing with significant results of the program, implications for research, and suggested additional studies.

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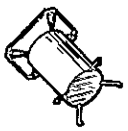

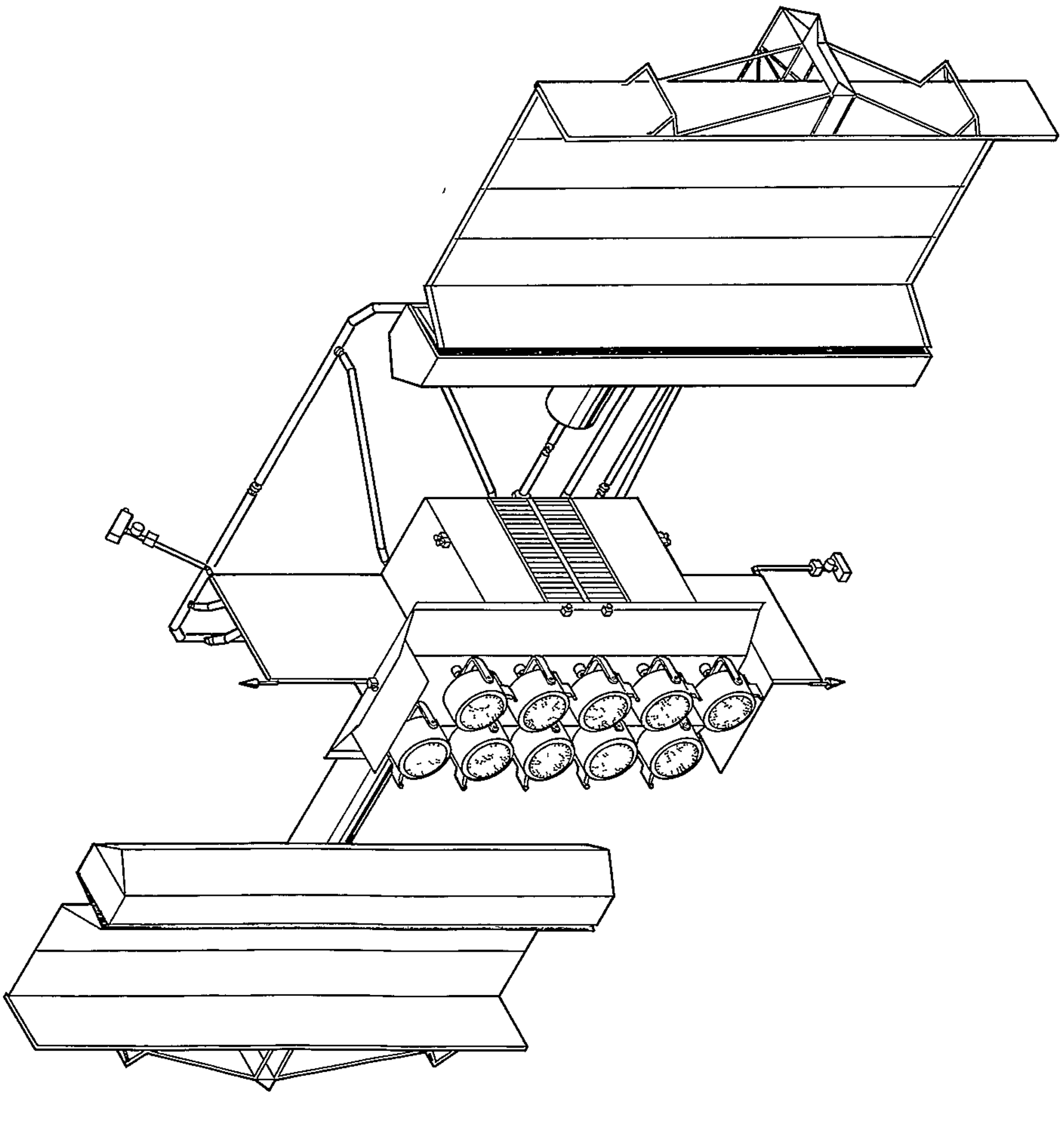

1 0 INTRODUCTION

This is a Phase A concept definition and system analysis study final report for a Solar Electric Propulsion Stage (SEPS) It was prepared for the Marshall Space Flight Center as the final submittal requirement under NASA Contract NAS8-30921 The program was ably directed by Contracting Office Representative Mr C H Guttman at MSFC A subcontract effort during this study phase was awarded to Hughes Aircraft Company to provide engineering data on ion thrusters The Hughes report is provided as an Appendix to Volume III of this report

Prior to the Phase A contract award both the Phase A and the follow-on Phase B efforts were negotiated It was originally planned that Phase B would be awarded approximately 2 months prior to Phase A completion Budget limitations precluded the Phase B award This revision to the plan also produced a change in the Phase A effort Planetary missions were emphasized during the first half of the study and a baseline configuration design was presented to MSFC at midterm At that time, direction was received to shift emphasis from planetary to Earth-orbital missions This necessitated repeating a similar design effort to develop a baseline Earth-orbital configuration and design for the final report Both baseline configurations, as well as discarded configurations, are addressed in Volume III of this report

1 1 PURPOSE

The Boeing objective for this study was to develop a versatile, reliable, and low cost SEPS concept Technical and configuration results developed would be available for input directly into the Phase B or payload utilization trade studies (PLUS) Fiscal funding and schedule requirements were to be established for a defined SEPS program Areas of concern that require new or continued development effort were to be identified

1 2 BACKGROUND

Electric propulsion has long been recognized as an efficient method for space propulsion To take advantage of this technique, electric propulsion development programs were initiated in the early 1960's and are continuing today The Space Electric Rocket Test I (SERT I) program was one of the first electric thruster flights performed in the mid 1960's The objective of this program was to prove that the thruster could start and operate in the space environment A follow-on program, SERT II, was for launch and life test of a 15-centimeter thruster Successful completion of these tests resulted in initiation of pre-phase A SEPS studies in 1971 These studies provided a preliminary definition of a SEPS system This Phase A study was directed toward systems analysis and design concept trades that would result in definition of a baseline configuration design

Certain primary ground rules and assumptions were well established The solar array was fixed at 25 kilowatts by existing AST programs, as were 30-centimeter thrusters With the change in emphasis from planetary to Earth-orbital operations, an Earth-orbital mission model was developed Requirements resulting from the model were used to develop the Earth-orbital baseline configuration design With this design, only minor changes are required to convert the stage to a planetary configuration

1 3 RESULTS

The general conclusion to be drawn from the study results is that a low cost, reliable Solar Electric Propulsion Stage can be designed that is readily adaptable to perform either Earth-orbital or planetary missions Further examination of both the hardware and operations are sure to provide technical and fiscal improvements prior to the technology cutoff date

From a programmatic viewpoint, the most significant results are that the schedule time of 52 months (from November 1976 to March 1981) is more than adequate for delivery of the first unit and that the total program cost is approxmiately $300 million Included in this cost is less than $100 million for DDT&E and less than $15 million each for 11 stages Development cost savings have been accrued by use of subsystems and hardware developed under the auspices of AST, Low Cost Systems office, and other space projects

Examples of further savings that could be realized are

1 Using a high-voltage solar array to eliminate the need for most of the power processor (estimated program savings of $23 million)
2 Developing a low-cost solar cell and cover (estimated savings of $2 million per stage)
3 Using the onboard computer for system test, thus eliminating much GSE (estimated program savings of $5-$10 million)

The major technical conclusion to be drawn from the study is that a SEPS can be designed that would accomplish all planetary missions in the mission model and provide improvement to the NASA Space Transportation System for delivery and retrieval of geosynchronous payloads For planetary missions, a probability of success of 0 9 is achievable for an Encke Rendezvous (single launch) For Earth-orbital operations, a longest single sortie probability of success of 0 97 can be attained

1 4 SCOPE

The remaining sections of this volume provide an overview of the Phase A study All references to cost will be general in nature With few exceptions, items discussed in this summary volume are discussed in more detail in the later volumes The most significant parts of this volume relative to further development of SEPS are Section 6 0 Significant Results, Section 7 0 Implications for Research, and Section 8 0 Suggested Additional Effort The remaining sections include discussions of study objectives (sec 2 0), method of approach (sec 3 0), study limitations (sec 4 0), and SEPS relationship to other NASA efforts (sec 5 0)

Cost analysis for the first half of the study was based on new development for all subsystems For the final report, the cost presented assumes subsystems are developed either through the Low Cost systems office efforts, AST efforts, or by other space programs The only subsystem development costs included are those related to SEPS-unique modifications

2 0 STUDY OBJECTIVES

The defined prime objective of the Phase A study activity is to provide sufficient data to initiate the SEPS Preliminary Design phase Ten specific ground rules constrained the limits of the study A program redirection at midterm changed the emphasis from planetary missions to Earth-orbital operation and impacted the configuration design by eliminating the use of the Titan 3E/Centaur launch vehicle and by changing the technology cut off date to September 1977

The first of the ground rules—maximum use of previous study results—led Boeing to the conclusion that its role in the study should be to develop a low cost, reliable, and versatile SEPS concept, which utilized, whenever practical, data and hardware previously developed For the midterm report, our objective was to provide a SEPS concept design approach During the final part of the contract, the Boeing goal was to develop a low cost design and program approach stressing Earth-orbital operations

These objectives were accomplished through the effort outlined within the six tasks defined in the statement of work Briefly, the tasks involved the following

1 Perform mission and system analyses to develop a SEPS concept design and development approach
2 Select and develop a SEPS concept design
3 Perform subsystem design, definition, and implementation analysis
4 Generate program plans and schedules
5 Perform cost analysis at program, system, and subsystem levels
6 Identify required supporting research and technology

3 0 METHOD OF APPROACH AND PRINCIPAL ASSUMPTIONS

The study plan was submitted with the SEPS proposal This plan outlined the effort required to perform the six tasks defined in the statement of work Major inputs, outputs, flow of tasks, and interrelationships between tasks are shown on the study logic diagram presented in figure 3 0-1 In the early stage of the contract, the evolutionary concept was eliminated and replaced with configuration and reliability trades As a result, mission and system analyses were performed concurrently with development of weight and reliability allocations These data were used in configuration design and subsystem trades that culminated in the planetary baseline configuration and subsystems recommended at midterm With redirection at midterm to emphasize Earth-orbital in lieu of planetary missions, it was necessary to repeat the earlier process for the new requirements Even though changes were made, the approach for the entire study was directed toward meeting the basic objectives described in section 2 0 of this document

During the performance of the study, the following primary assumptions applied

1 Initially the first flight of the SEPS was assumed to be an out-of-ecliptic mission or Comet Encke Slow Flyby in 1979 This was changed during the course of the study to an Encke Rendezvous mission for first flight in 1981 followed by an Earth-orbital test flight in the same year
2 The Tital III E/Centaur was assumed as the launch vehicle for the initial study phase, with the Shuttle/IUS or Tug emphasized for the latter half of the study
3 Use of the DSN system as of 1977 was assumed for planetary missions, while the STDN was used for Earth-orbital missions
4 Solar array power of 25 kilowatts (BOL) was assumed for the entire study
5 Eight-thruster operation with 10,000-hour lifetime (full power) was considered for the first half of the study, with the lifetime extended to 20,000 hours for the latter half The random failure rate did not change
6 5-year lifetime including coast periods, was used for the equipment
7 SEPS Earth-orbital operation was constrained to 13 334 kilometers (7200 nmi) altitude and higher
8 SEPS Earth-orbital operations considered mercury propellant loading optimized for each sortie with refueling accomplished via the Tug vehicle
9 SEPS was capable of multiple rendezvous and docking maneuvers and provided for payload exchange with the Tug vehicle
10 The launch vehicle for GEOSEPS consisted of the Shuttle and a SEPS-optimized cryogenic Tug
11 Payload characteristics are in accordance with the latest Shuttle payloads definition
12 The SEPS Earth-orbital traffic model will accommodate all geosynchronous missions specified in the January 1974 Space Shuttle Traffic Model

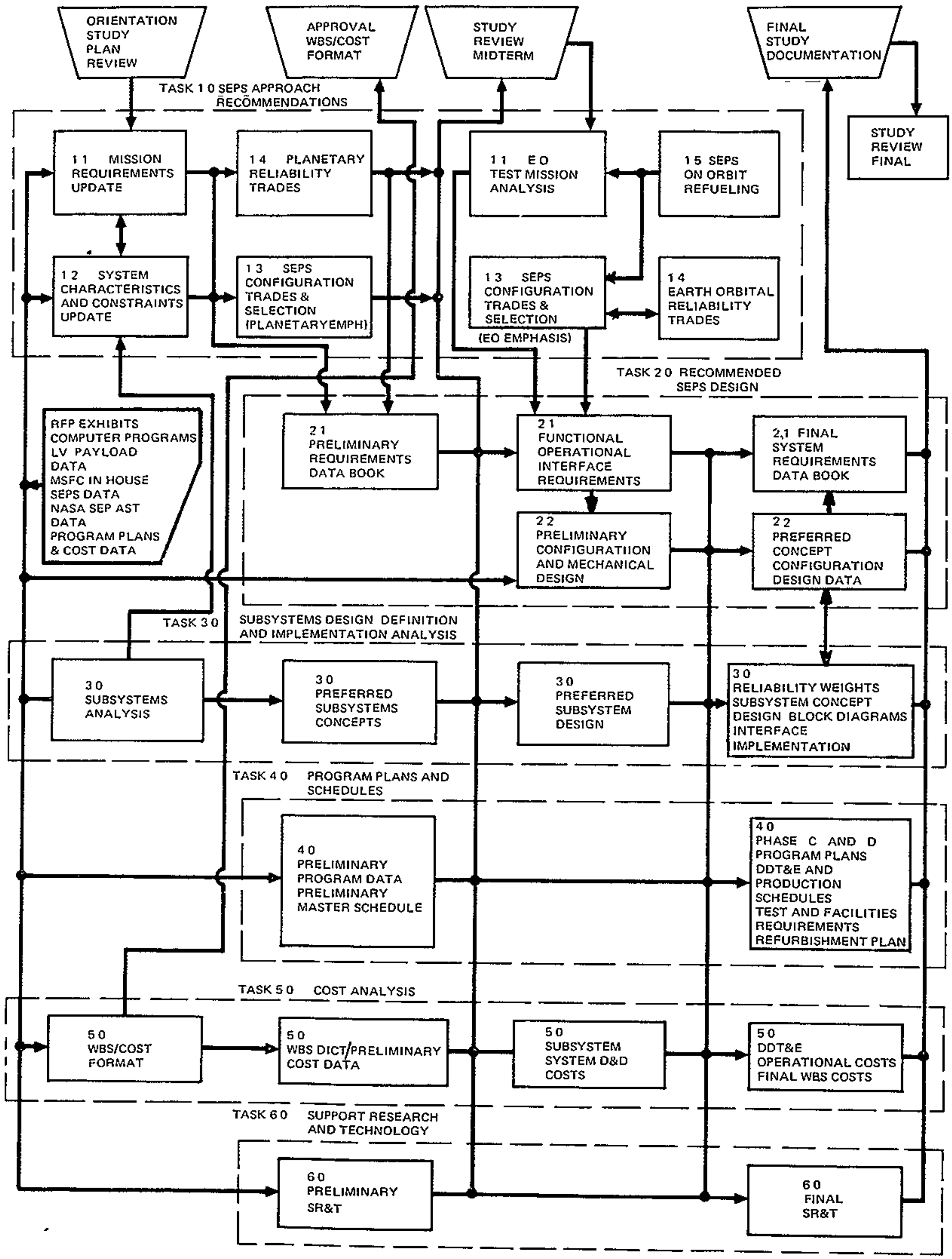

Figure 3 0-1 Study Logic Diagram

4 0 STUDY LIMITATIONS

The study ground rules define most limitations for this study In this stage of a program, however, many of the interfacing systems are continually changing This study was necessarily limited by the point at which these systems were fixed Also, the change in study emphasis from planetary missions prior to midterm to Earth-orbital missions resulted in some ground rules or assumptions not receiving the level of consideration desired Where more detail was felt to be necessary, it is provided in this section

4 1 PAYLOAD DEFINITION

In performing the payload traffic model analysis and developing the stage configuration design, a number of assumptions were necessary relative to payload definition Payload outside dimensional size assumed that all interface requirements would be accommodated in the payload design Docking assumed identical docking fixtures for all payloads Payload power and fluid requirements from the stage were assumed to be satisfactory The transporting of multiple payloads to geosynchronous orbit and return was assumed practical

4 2 SOLAR ARRAY

Constant solar cell efficiency and negligible degradation were assumed This seemed reasonable in light of improvements in cell technology projected for the 1977-1978 time period Reliability analysis assumed the information provided by Lockheed was accurate Dynamic characteristics for these large arrays were assumed to be satisfactory This, however, demands careful scrutiny

4 3 STAGE STABILITY

Payload stack dynamics analyses were not accomplished in this study It was assumed that neither rigid nor flexible body dynamics would cause weight or control problems Future studies should verify structure and control system adequacy for various payload stacks during boost, docking, and payload interchange with the Tug

4 4 REFURBISHMENT

It was initially assumed that refurbishment would not be required for the missions planned Allowing additional trip time for payload delivery due to reduced propulsion capability was assumed to be an acceptable backup mode of operation Therefore, refurbishment received only a cursory analysis Additional study is required to trade the advantages and disadvantages of refurbishment at various points in stage life

5 0 RELATIONSHIP TO OTHER NASA EFFORTS

In general, Boeing utilized the Solar Electric Propulsion-advanced system technology (AST) for the Earth-orbital and planetary performance predictions and configuration synthesis Table 5 0-1 summarizes the data topic, source, and level of utilization Technology data were obtained as government-furnished data (GDF), including the precursor MSFC Phase A studies (and implicitly, the AST used therein) and SEPS-peculiar hardware development data from LeRC, Hughes, JPL, and Lockheed Whereas the GFD provided useful historical information, the data utilized directly in the Boeing studies were obtained firsthand through a Hughes subcontract and via personal contacts with other SEPS-related organizations This process was used because it was quickly determined that technology development is in a vigorous stage and additional AST will be required to ensure SEPS feasibility

Considerable trajectory data for electric propulsion has been computed as part of the SEP-AST work, this data was utilized during this contract to minimize new trajectory synthesis time One result of the Boeing contract work is determination that additional mission design must be accomplished based on some value of thruster efficiency representative of a minimum statistical bias This means that planetary missions must be designed such that the nominal missions will have a mandatory terminal coast maneuver In case of low propulsion performance, this terminal coast is utilized to make up the required impulse with available flight performance reserves

Table 5 0-1 SEP-AST Utilization

SEP-AST	Data source	Utilization
Ion thruster	Hughes	Efficiency, weight, dispersions, cost, envelope, and interfaces per Hughes recommendation
Power processing unit	Hughes	As above Boeing designed thermal control
Propellant storage and delivery	JPL-Hughes —LeRC	Adopted JPL storage and latching valves Boeing designed delivery and damping
Solar array	Lockheed	Adopted directly Revised weights per Boeing analysis Recommended a low cost manufacturing procedure
TVC	JPL-Rockwell	Rejected Boeing concept is simpler, more flexible design
Spares switch	JPL	Switch design adopted Interface concept selected by Boeing
Thruster array	JPL-Rockwell	Rejected Boeing optimized array selected

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6 0 BASIC DATA GENERATED AND SIGNIFICANT RESULTS

As technology of electrical propulsion has been maturing, so has interest been quickening in utilizing this technology for the fulfillment of meaningful space missions Initial system studies at the Jet Propulsion Laboratory (JPL) showed the desirability of using ion thrusters powered by large solar arrays for accomplishing deep-space missions Subsequent studies by Rockwell International under the sponsorship of NASA's Marshall Space Flight Center (MSFC) have reaffirmed the capability of an electrically propelled vehicle to achieve significantly higher performance on planetary missions More recently, continuations of these studies for MSFC have indicated that a cost and performance benefit may be realized by the inclusion of this technology in the upcoming (Shuttle-based) Earth-orbital transportation system The Earth-orbital traffic analysis studies presented in Volume II of this report (and summarized in this section) verify that a vehicle employing solar electric propulsion technology is a cost-effective method for boosting massive payloads to the higher Earth orbits

The present study represents a further extension of the earlier investigations into the applications of electric propulsion to both deep-space missions and to geocentric transportation roles For this study, the high-technology vehicle takes the form of a Solar Electric Propulsion Stage (SEPS) This stage represents a standardized design to accomplish the multitude of missions for which the technology is suitable This concept is not new for this study, and many facets of the mission and stage design are drawn directly from, or represent direct extensions of, the design analyses carried out in the previous studies by JPL and by Rockwell However, in other areas, the earlier designs were discarded as inappropriate or outdated by recent technology developments The assessment of the state of NASA's Advanced System Technology (AST) development program, and the utilization of data therefrom, has been an integral part of this study program

Figures 6 0-1 and 6 0-2 show the selected SEP stage adapted respectively for Earth-orbital operation and planetary use (a Comet Encke rendezvous) This vehicle is designed to satisfy the requirements and objectives as provided by NASA/MSFC

The stage ion propulsion and large solar array shown are based upon the ongoing NASA advanced system technology program All of the remaining stage systems have either flown before or are adapted from other space systems By this means, the SEP stage development risk is minimized

The introduction of heat pipe power processor cooling by NASA allows for many alternative stage configurations, which also tends to reduce risk Figure 6 0-3 shows four stage central bus structure alternatives Final selection of a recommended design (shown on figs 6 0-1 and 6 0-2) was similar to the design shown in the upper right corner of the figure Each of the alternatives shown was designed to investigate a different load path and structure concept The upper left design turned out lightest by a small amount but the upper right was shortest Since length was more serious problem in the Shuttle than weight, a design similar to the upper right was selected

The remainder of this section will discuss the conclusions reached in the system and mission analysis and the subsystem analysis areas

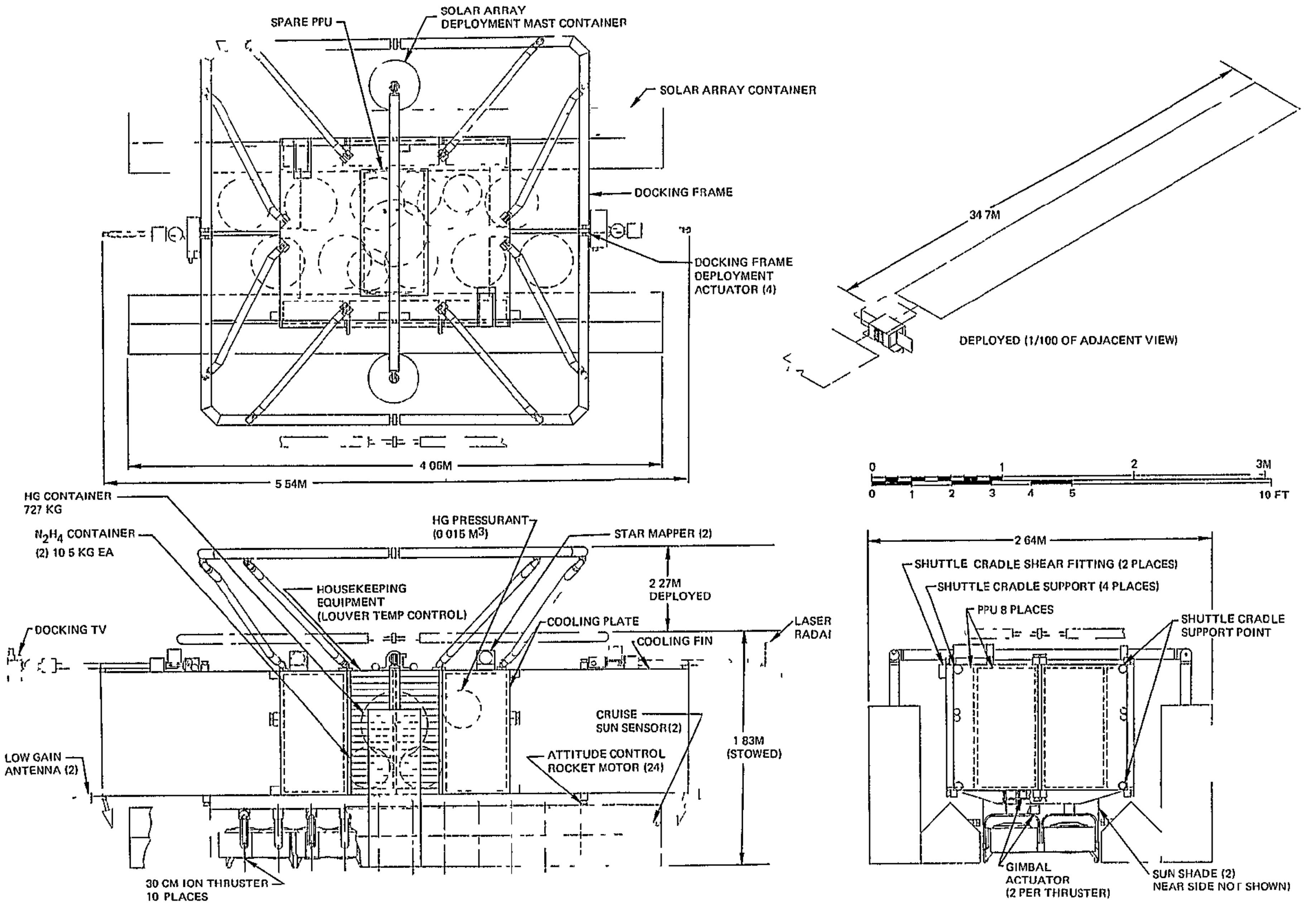

Figure 6 0 1 General Arrangement – SEPS Model 1038 7/EO

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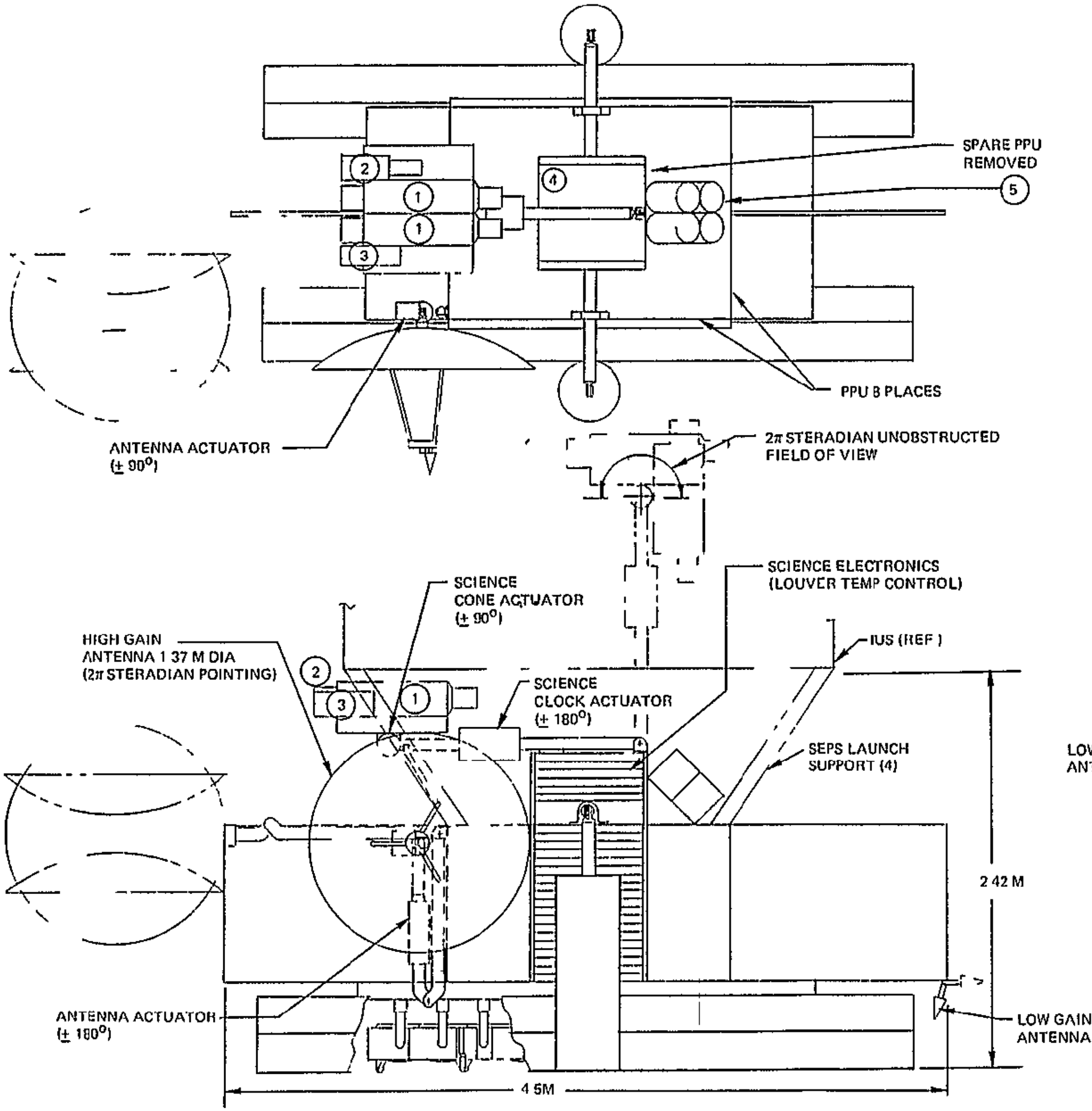
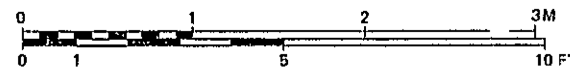


SCIENCE INSTRUMENT FIND NUMBERS

1 TELEVISION
2 IR RADIOMETER
3 UV SPECTROMETER
4 MASS SPECTROMETER
5 DUST DETECTOR



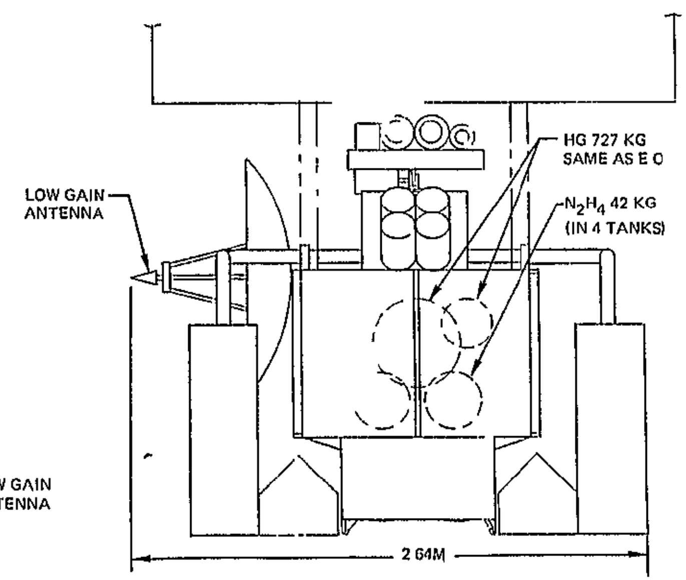


Figure 6 0 2 General Arrangement — SEPS Model 1038 7/PLAN

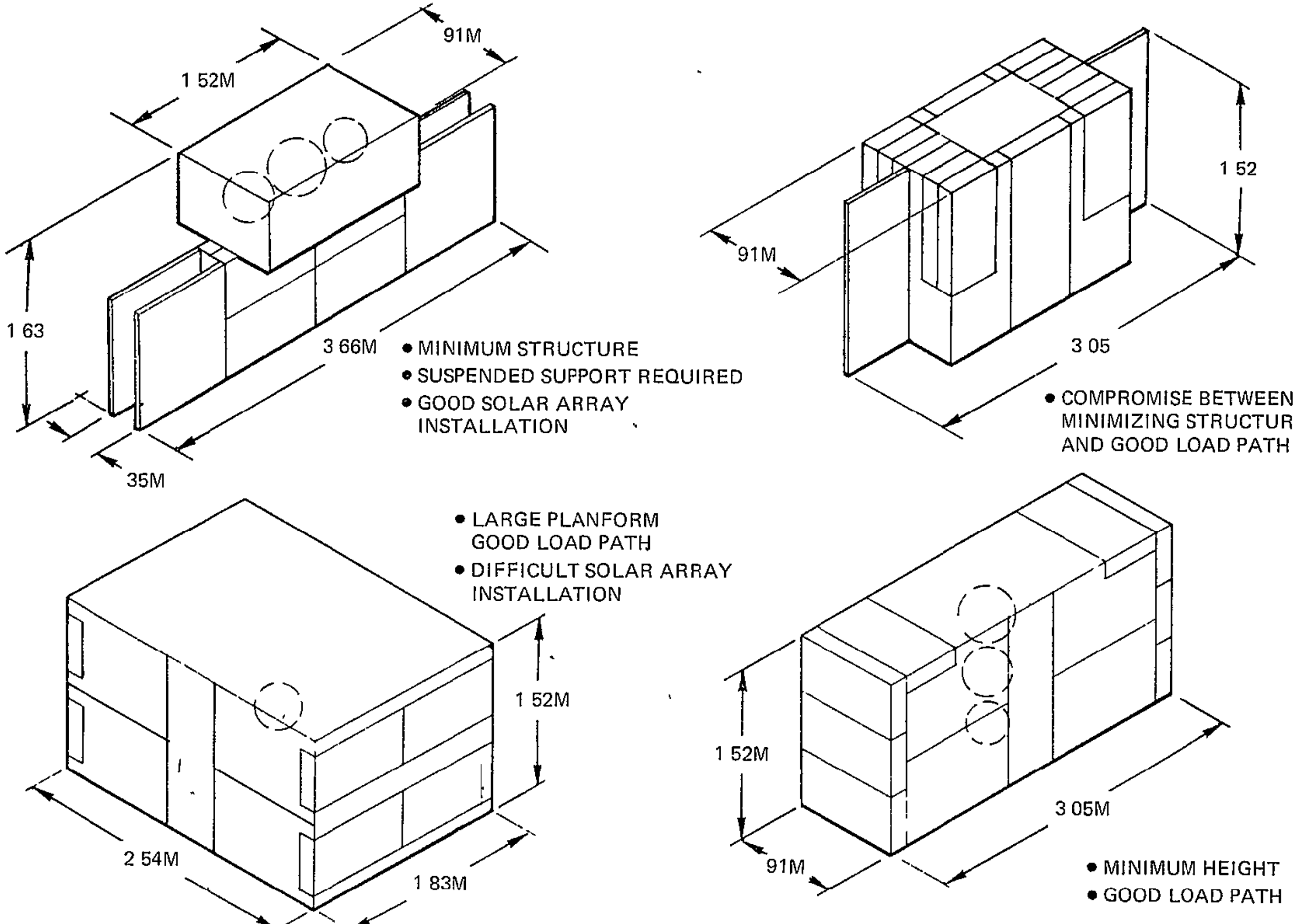


Figure 6 0-3 SEPS Structural Arrangement Alternatives

6 1 MISSION AND SYSTEM ANALYSIS

Detailed mission and system analyses, as documented in Volume II, have been conducted on the application of SEPS to the candidate missions These analyses were conducted within the framework of study groundrules and constraints previously discussed The following discussions highlight the major results of these analyses

6 1 1 Earth Orbital Missions

The earth orbital mission model includes an Earth Orbital Test mission in 1981 followed by initiation of operational Geosynchronous Transportation missions in 1982

SEPS Performance in Earth Orbit—SEPS performance capability for transfer to synchronous orbit, longitude phasing, and plane change are shown on figures 6 1-1 and 6 1-2 Study results have shown that operation of SEPS from a relatively high (>13,300 km) IUS/Tug payload transfer (changeover) orbit is required to (1) minimize solar array radiation degradation and (2) shorten trip times thereby accommodating the current geosynchronous payload traffic model

SEPS Traffic Analysis—To determine both the adequacy of SEPS performance in Earth orbit and the characteristics and requirements for the Earth-orbital test mission, a traffic analysis was conducted comparing the Shuttle/Tug/SEPS to the Shuttle/Tug alone A basic premise of the analysis was to maximize SEPS/Tug use to minimize Shuttle flights and operation, the goal being that the total cost of developing and deploying the SEPS system would, as a minimum, be offset by the savings made in the Shuttle program The analysis was initiated by examining the January 1974 Space Shuttle Traffic Model and updating it to accommodate the more recent July 1974 payload descriptions An initial traffic model, utilizing SEPS and the baseline 9 14-meter (30-ft) Tug was then developed This model showed little savings over the Shuttle/Tug model due to payload delivery limitations imposed by the 9 14 meter length of the baseline Tug To take maximum advantage of SEPS, it is necessary to deliver as many payloads as practical on each sortie To overcome the payload quantity limitation with the Tug, an optimized Mini-Tug was defined that provided an additional 1 52 meters (5 ft) of payload space in the Shuttle bay The SEPS traffic model was then updated using this Mini-Tug The resulting model, showing individual sortie requirements, is illustrated in figure 6 1-3 The model requires four operational SEPS with utilization factors (in terms of burn life) as shown in table 6 1-1

The traffic model is sensitive to SEPS/payload weight Any weight increase ricochets through both the SEPS and the Tug Increased payload requires more SEPS propellant, which requires more flight time, and may result in a lower changeover altitude, which requires more propellant, which again requires more flight time Increased single sortie time may reduce the total number of sorties, thus requiring additional SEPS to support the model Any decrease in Tug capability or SEPS total thrust (due to solar-array degradation) will also adversely affect the traffic model

The model is less sensitive to engine burn time, reliability, stage lifetime, and other features of the SEPS, since these do not have the similar compounding features of a weight increase The critical point in the model is keeping the average Tug/SEPS changeover altitude as high as possible (≈ 18,000 km) This provides good SEPS utilization and a Tug with capability to support other low Earth-orbital flights

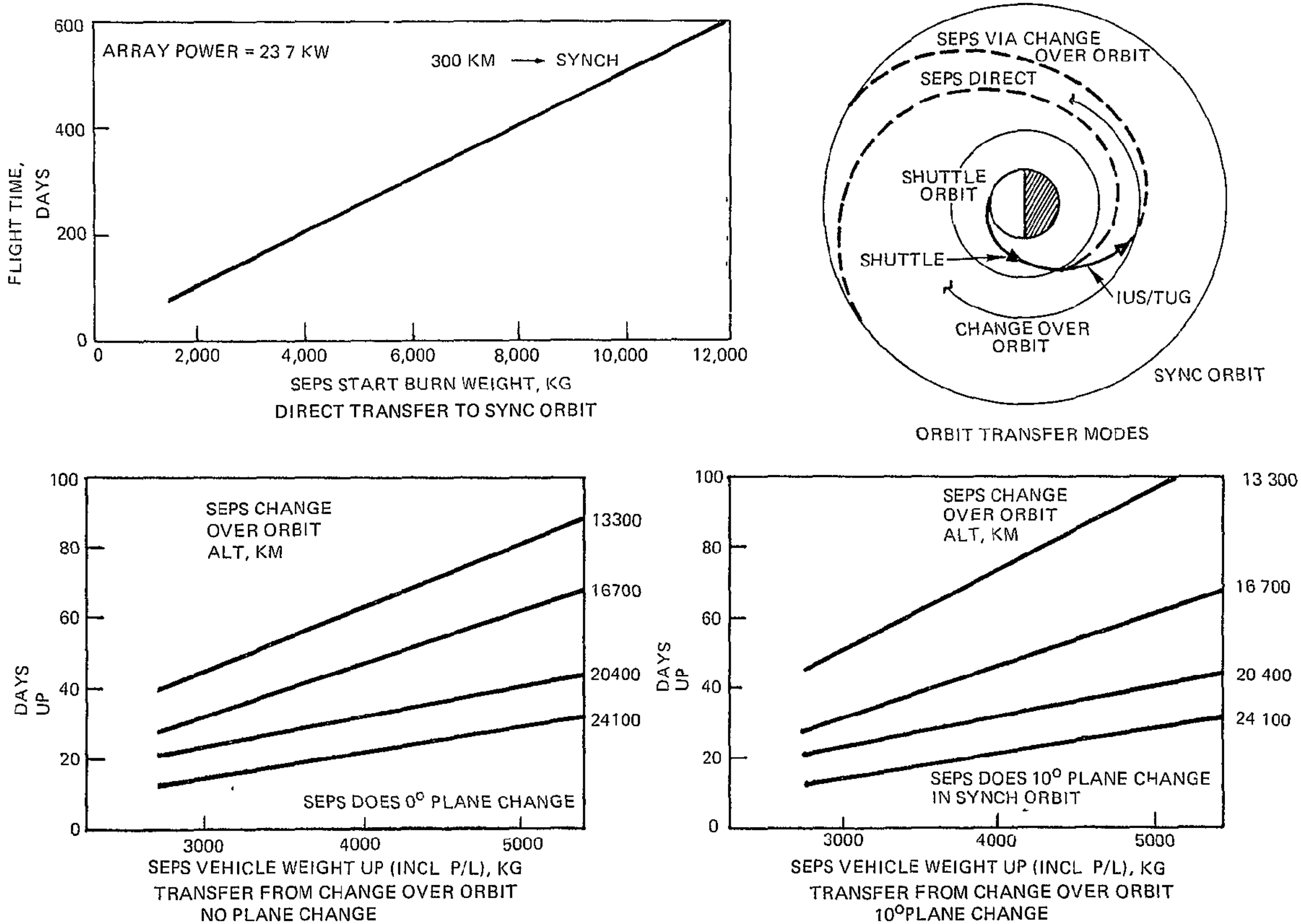


Figure 6 1—1 SEPS Orbital Performance—Orbit Transfer

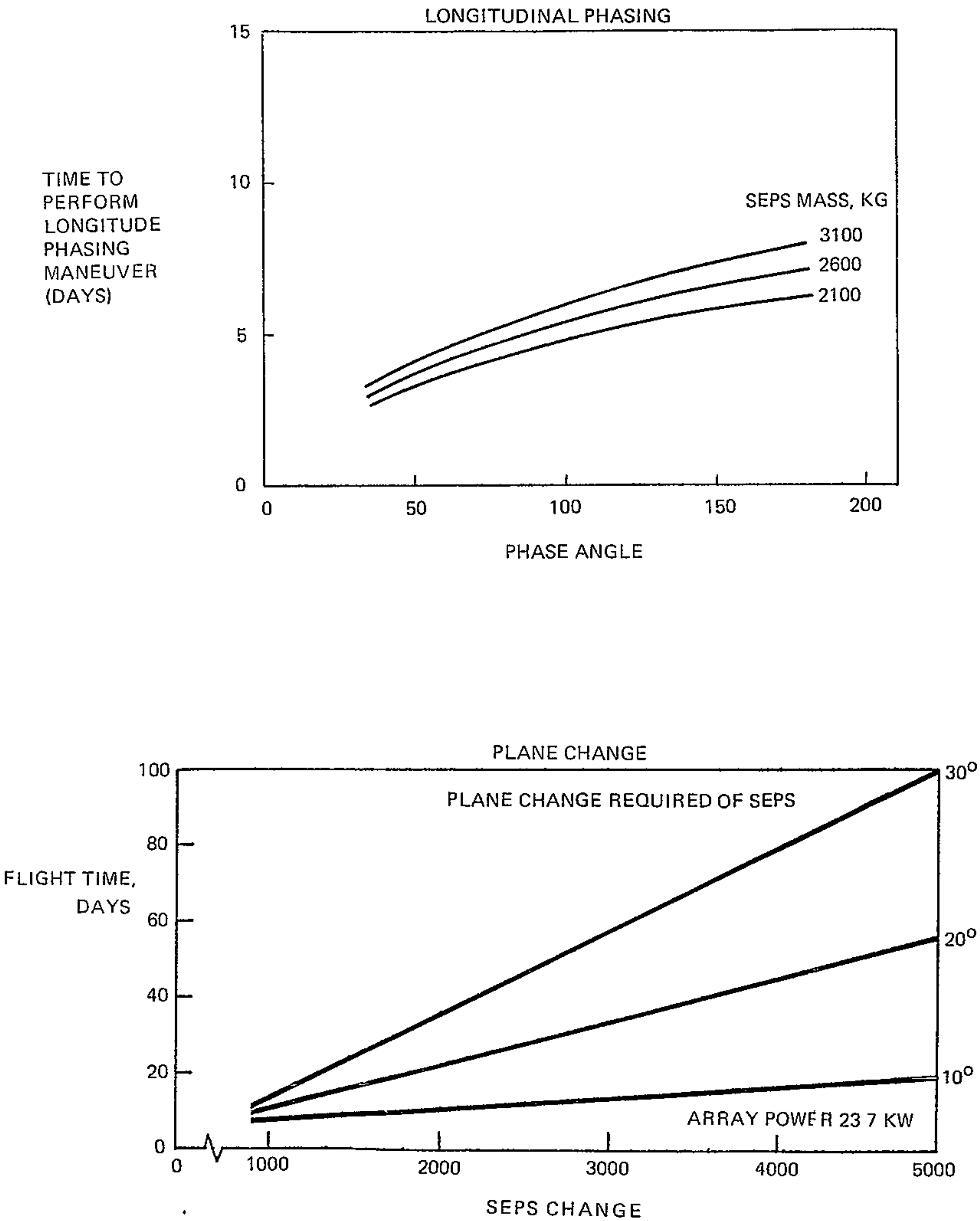


Figure 6 1-2 SEPS Earth Orbital Performance
-Orbital Phasing and Plane Change-

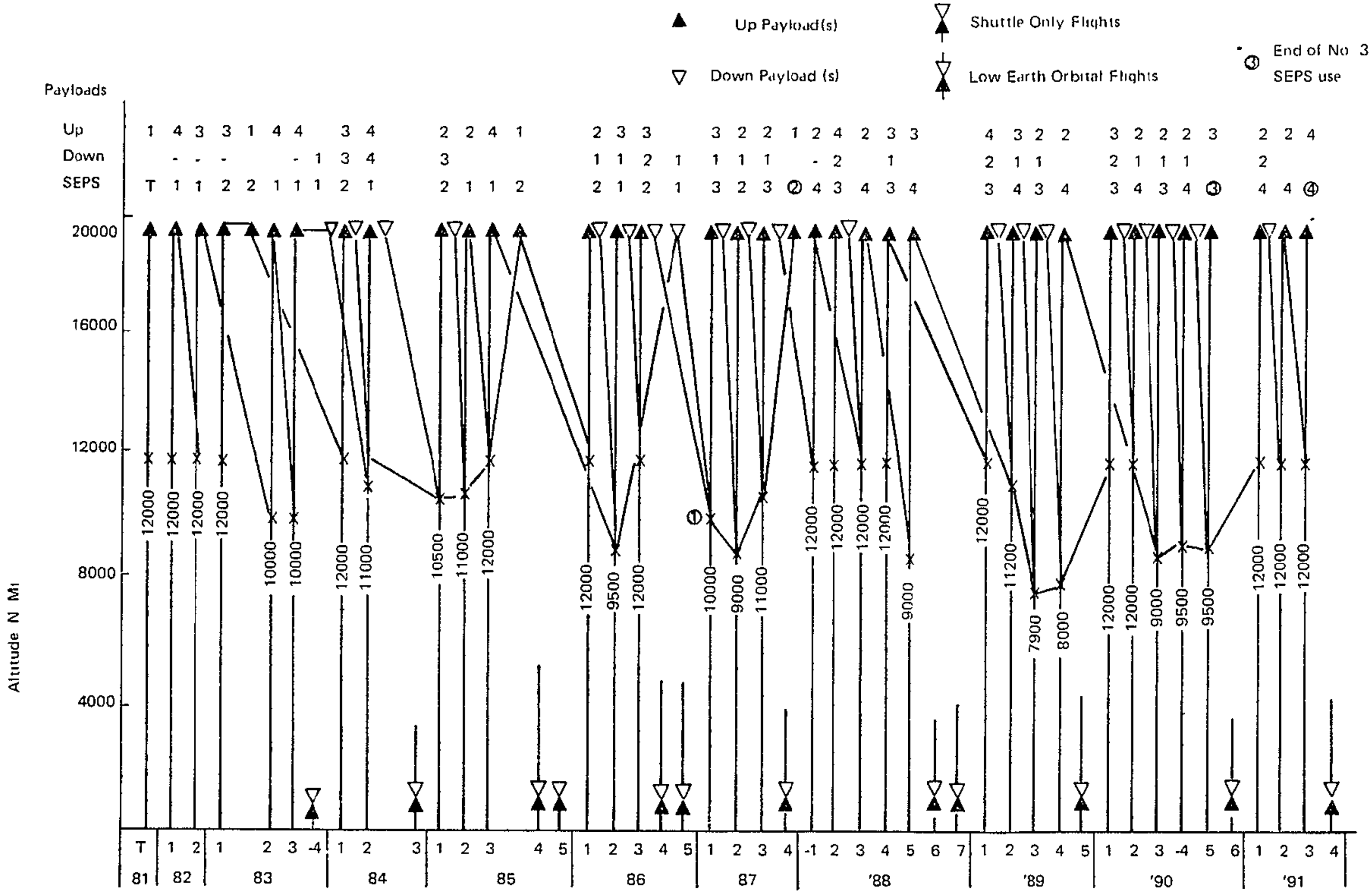


Figure 6 1-3 SEPS Traffic Model Sortie Summary

Table 6 1-1 Traffic Model Burn Time Summary

SEPS Vehicles	81	82	'83	84	85	86	87	88	89	'90	91	Total ▷1
T	68 (1)											68 (1)
1		95 (2)	193 (½)	107 (½)	134 (2)	136 (2)						665
2			95 (1)	99 (1)	144 (2)	172 (2)	120 (2)					630 8)
3							176 (2)	177 (2)	229 (2)	245 (3)		827 (9
4								204 (3)	207 (2)	197 (2)	197 (3)	805 (10)

(2) Sorties per year
Longest mission 89 3 at 136 days

▷1 Total engine burn days 20000 hr engines = 833 day
8 of 10 engines operating = 1040 day of SEPS operation

SEPS Life Is 5 Years

Table 6 1-2 shows the impact of the proposed traffic model on the Shuttle program From cursory comparisons, the SEPS program will pay its way (savings equivalent to additional expense), but there appears to be little potential for a significant cost savings in the multibillion-dollar Shuttle program The Mini-Tug, optimized for combined SEPS/Shuttle use, would probably show a savings over the proposed baseline Tug On the other hand (and perhaps more promising), the candidate geosynchronous payloads could be redefined to optimize their volumetric efficiency for the SEPS/Tug application In this case, the SEPS/Mini-Tug model would apply to the baseline tug also

The SEPS could add a high degree of flexibility to the Shuttle program A backup vehicle would provide full mission redundancy, down traffic could be increased to either clear particular orbit areas or recover additional payloads The SEPS could recover spent payloads and position them in a noninterference orbits

Should the Shuttle program be limited to use of a reduced capability IUS (transtage, solid, etc), the SEPS role can still be valuable in positioning multiple payload launches in geosyrchronous orbit A solid Tug could put more payloads at changeover altitude than it could place in geosynchronous orbit The SEPS could then place and move these payloads in geosynchronous orbit Assuming an IUS with performance capability of the Mini-Tug, the SEPS sorties would be the

Table 6 1-2 SEPS Impact on 10-Year Traffic Model

	SHUTTLE 30' TUG	SHUTTLE 30' TUG & SEPS	SHUTTLE MINI-TUG & SEPS
SHUTTLE FLIGHTS	69	62	46 [1]
TUGS			
STRETCHED TRANSTAGE	13	8	6
TUG	56	52	0
MINI-TUG	–	–	38
SEPS [2]	0	8	5
PAYLOADS UP	126	126	126
GEO/SYNC	(96)		
LOW E/O	(30)		
PAYLOADS DOWN	69	69	69
GEO SYNC	(33)		
LOW E/O	(36)		

[1] ELEVEN ADDITIONAL SHUTTLE FLIGHTS COULD BE SAVED BY COMBINING THE LOW E/O PAYLOADS IN THE BALANCE OF THE SHUTTLE PROGRAM

[2] INCLUDES THE 1981 SEPS TEST FLIGHT

same, and the change in the model would be to eliminate the SEPS and Tug down payloads and raise the changeover altitude accordingly IUS's of lesser capability could be used and a model determined using additional (more than four) SEPS vehicles

Earth-Orbital Test Mission—The Earth Orbital Test (EOT) Mission has as its basic objective the demonstration of critical functions associated with the operational SEPS geosynchronous transportation mission

The ability to demonstrate these mission functions by way of the Earth-orbital test is primarily dependent upon the EOT launch vehicle and allowable mission cost With the Shuttle/IUS specified for this mission, the following flight objectives are proposed

1 Demonstrate deployment of SEPS/payload at changeover orbit
2 Demonstrate ascent to synchronous orbit
3 Demonstrate multiple payload deployments and retrievals
4 Demonstrate SEPS orbital phase and plane change capability
5 Demonstrate SEPS descent to changeover orbit
6 Accommodate, as a contingency, descent to Shuttle orbit for SEPS recovery

The SEPS configuration for the EOT mission is assumed to be identical to that for subsequent operational missions The EOT mission is currently scheduled for launch in 1981, with operational missions starting in 1982

The test payload selection is optional at this time Candidate approaches are (1) development of a new test vehicle, (2) use of a scheduled operational spacecraft, or (3) use of residual or test hardware from a previous or concurrent spacecraft program The primary requirement on the payload is that it interface as required with SEPS

Preliminary weight allocations have been made for the EOT mission as follows

SEPS (less Hg)	1 304 kg (3,007 lbm)
Mercury propellant	813 kg (1,792 lbm)
Test payload	907 kg (2,000 lbm)
Total	3 084 kg (6,799 lbm)

Figure 6 1-4 is a pictorial representation of the candidate EOT mission Launch is due east from Kennedy Space Center via the Shuttle /IUS The Shuttle is separated at an altitude of 300 kilometers (160 nmi) The IUS provides the energy required to transfer from the 300-kilometer Shuttle orbit to a circular changeover orbit at 22 200 kilometers (12,000 nmi) This is accomplished with a perigee and apogee thrust maneuver Each maneuver includes thrusting in the yaw plane to accomplish an overall 18 5-degree plane change The SEPS then ascends to a 10-degree synchronous orbit and deploys the test spacecraft To demonstrate orbital phasing maneuvers required for multiple payload deployment/retrieval, the SEPS performs a lead maneuver placing it 60 degrees ahead of the payload A subsequent 60-degree lag maneuver is followed by rendezvous and docking with the test payload SEPS again deploys the payload and accomplishes a 10-degree plane change to obtain an equatorial synchronous orbit A second plane change maneuver reestablishes the initial 10-degree inclination synchronous orbit, where a second rendezvous and docking with the payload

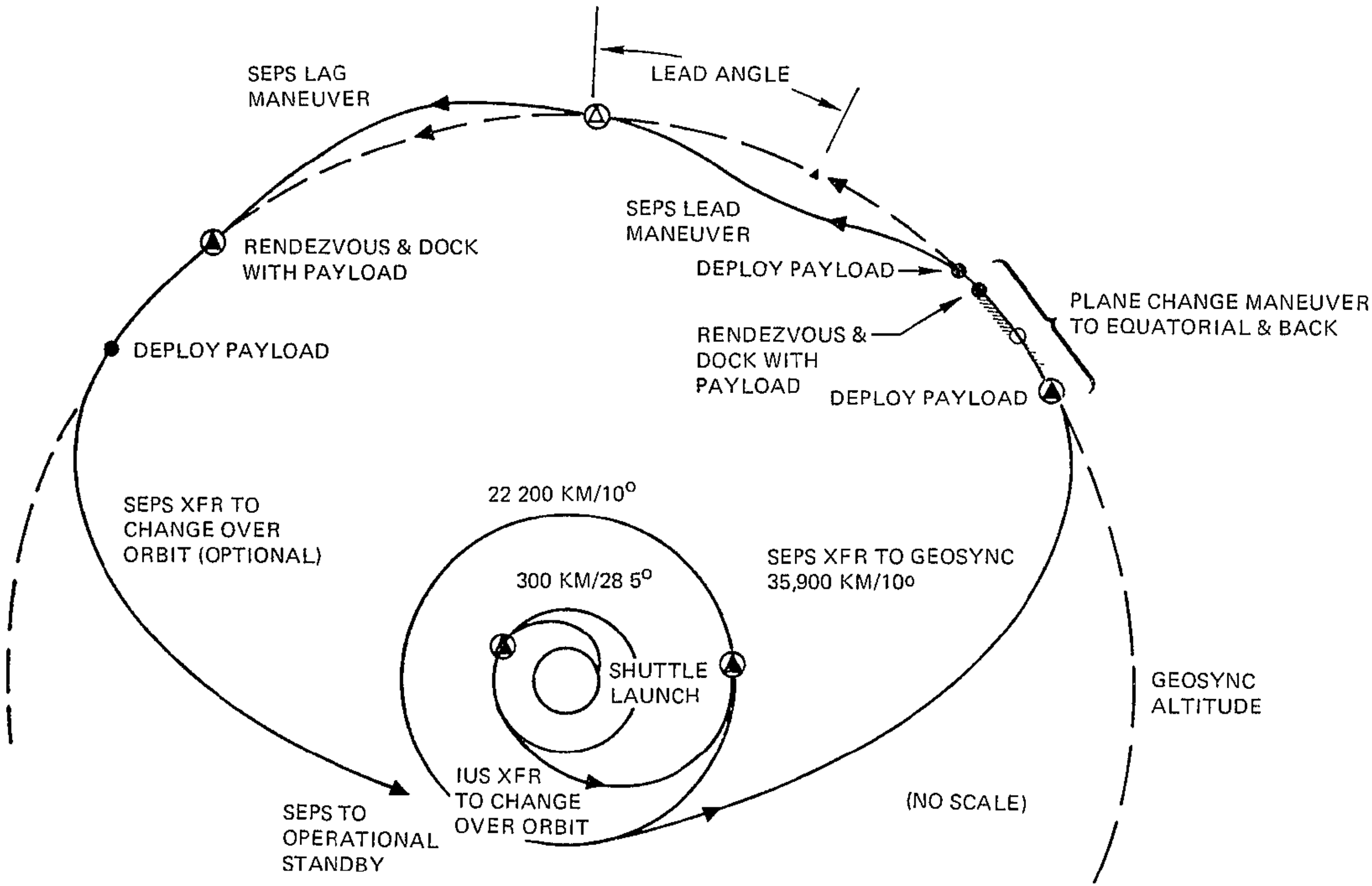


Figure 6 1-4 Candidate Earth Orbital Test Mission Profile is accomplished The payload is deployed and left in synchronous orbit while the SEPS descends to the changeover orbit At this point, the SEPS demonstration flight is complete Alternatives at termination include shutdown, transfer to operational standby, or descent to Shuttle parking orbit for recovery Since the SEPS for the Earth-orbital Test Mission is identical to the operational vehicles, it appears prudent from a cost standpoint to maintain its operational availability in orbit While this vehicle has not been integrated into the traffic analysis, this option has been selected as the baseline mode In the absence of test mission failures or problems, the recovery of SEPS does not appear warranted Radiation damage in traversing the radiation belts down to the Shuttle pickup orbit could obviate meaningful ground evaluation or result in increased costs to refurbish for subsequent use

A contingency mode has been included to facilitate SEPS recovery by the Shuttle should serious test failures occur in payload-associated activities such as deployment and retrieval and where visual ground inspection is necessary Prior to descent, a plane change maneuver is required to achieve the Shuttle orbit inclination of 28 5 degrees The operational mission sequence for this alternate and for the baseline Earth Orbital Test Mission is shown in figure 6 1-5 The time required to accomplish each of the maneuvers is indicated In the event of a failure and selection of the contingency mode, more mercury propellant is consumed, and the mission requires more time

Mercury propellant for the baseline mission is 141 kilograms and for the contingency mission 560 kilograms, well within the maximum mercury load of 813 kilograms allocated The excess propellant can be used to accommodate (1) weight growth in the test vehicle, (2) flexibility in test payload selection, or (3) as additional mercury to support operational SEPS missions

The major concern relative to this test mission is its close proximity to the first operational mission The current mission model and program schedule provides only 7 months from end of EOT mission to first operational launch The timeliness of test results is questionable It is recommended that the mission be rescheduled to precede the first operational mission by at least 12 months The current development schedule will support the earlier launch An earlier launch could also provide SEPS engineering verification prior to the Encke Rendezvous mission

6 1 2 Planetary Missions

The planetary mission model for this SEPS study includes the following missions (1) 1981 Encke Rendezvous, (2) 1982 Mariner Jupiter Orbiter, (3) Venus Radar Mapper, (4) 1984 Pioneer Jupiter Probe, (5) 1985 Saturn Orbiter/Probe, (6) 1986 Metis Rendezvous, and (7) 1987 Mercury Orbiter Each of these missions has been examined in detail so that an optimized baseline trajectory can be selected This baseline trajectory then provides the basis for the design of the various subsystems Important mission characteristics that have resulted from this trajectory design process are summarized in table 6 1-3 This table identifies the more significant trajectory parameters and the resultant mass properties Where particular trajectory characteristics tended to drive a subsystem design, the trajectory was analyzed to determine if modifications could be made to ease the subsystem design requirement without severely compromising the optimized trajectory For example, all trajectories are mass optimized except for the 1987 Mercury Orbiter In the case of Mercury, it was found that a mass optimized trajectory yielded propulsion burn times on the order of 450 days (eight-thruster continuous full power operation) When reliability analyses indicated unacceptably low probability of mission success, the trajectory was modified to reduce burn time to about 315 days

OPERATION	TIME, DAYS
SHUTTLE LAUNCH TO 300 KM CIRC ORBIT @ INCL = 28 5°	0 → 0 5
IUS (STRETCHED TRANSTAGE) TRANSFER FROM 300 KM TO 22,200 KM CIRC @ INCL = 10°	0 5 → 1 0
SEPS TRANSFER FROM 22,200 KM TO SYNCH ALT @ INCL = 10°	1 0 → 24 5
SEPS ORBITAL OPERATIONS	
DEPLOY TEST PAYLOAD	24 5 → 25
SEPS LEAD MANEUVER OF 60°	25 → 29
SEPS LAG MANEUVER OF 60°	29 → 33
RENDEZVOUS & DOCK WITH PAYLOAD	33 → 35

↓

DISCARD PAYLOAD

OR

CONTINGENCY OPERATION	TIME
SEPS TRANSFER FROM 10° INCL TO 28 5° INCL	35 → 55
SEPS TRANSFER FROM SYNCH ALT TO 300 KM	55 → 200

BASELINE OPERATION	TIME
SEPS TRANSFER FROM 10° INCL TO 0° INCL	35 → 40
SEPS TRANSFER FROM 0° INCL TO 10° INCL	40 → 45
SEPS TRANSFER FROM SYNCH ALT TO 22,200 KM	45 → 52
RENDEZVOUS WITH IUS IF DESIRED	

Figure 6 1-5 Earth Orbital Test Mission Time Sequence

Table 6 1-3 SEPS Planetary Mission Characteristics

PARAMETER	1981 ENCKE RENDEZVOUS	1982 MARINER JUPITER ORB	1983 VENUS RADAR MAPPER	1984 PIONEER JUPITER PROBE	1985 SATURN ORB PROBE	1986 METIS RENDEZVOUS	1987 MERCURY ORBITER
LAUNCH VEHICLE*	SHUTTLE/IUS	SHUTTLE/IUS/5KS	SHUTTLE/IUS	SHUTTLE/TUG/TKS	SHUTTLE/TUG/TKS	SHUTTLE/TUG	SHUTTLE/TUG
LAUNCH ENERGY, C_3 (KM^2/SEC^2)	20 25	28 1	4 0	36 0	50 4	9 0	16 0
LAUNCH DATE	7 MAR 81	16 JAN '83	5 JUNE 83	10 FEB 84	31 DEC 84	14 JULY '86	25 MAY 87
FLIGHT TIME TO TARGET (DAYS)	1045	800	130	800	1800	600	400
PROPULSION DURATION (DAYS)	1005	329	123	394	429	600	316
STAGE/PAYLOAD SEP (DAYS FROM LAUNCH)	NA	329	130	394	429	NA	400
TARGET ARRIVAL DATE	16 JAN 84	26 MAR 85	13 OCT 83	20 APRIL 86	5 DEC 89	5 MAR 88	28 JUNE 88
STAGE SOLAR DISTANCE (AU) MAXIMUM	3 7	3 2	1 0	3 6	4 5	2 5	1 0
MINIMUM	1 0	1 0	0 72	1 0	1 0	1 0	0 4
STAGE/PAYLOAD SEPARATION	NA	3 2	0 72	3 6	4 5	NA	0 4
TARGET ARRIVAL	1 7	NA	(0 72)	NA	NA	2 5	(0 4)
STAGE COMMUNICATION DISTANCE (AU)							
MAXIMUM	4 5	3 8	0 51	4 3	4 8	3 3	1 7
STAGE/PAYLOAD SEPARATION	NA	3 8	0 51	4 3	4 6	NA	1 3
TARGET ARRIVAL	1 4	NA	(0 51)	NA	NA	1 9	(1 3)
TARGET APPROACH VELOCITY, VHP (KM/SEC)	0	7 0	4 0	6 7	6 0	0	2 0
PAYLOAD RETRO ΔV (KM/SEC)	NA	1 53	3 17	NA	1 35	NA	0 53
CAPTURE ORBIT RADIUS (R/R_O)	NA	(3 x 60)	2 0	NA	(3 x 60)	NA	1 5
PAYLOAD TYPE/DERIVATION	ATTACHED SCIENCE PKG /NEW	SEPARABLE SPACECRAFT /MJS	SEPARABLE SPACECRAFT /MARINER	SEPARABLE SPACECRAFT /PIO	SEPARABLE SPACECRAFT /MJS	ATTACHED SCIENCE PKG /NEW	SEPARABLE SPACECRAFT /VIKING
MASS PROPERTIES (KG)							
PAYLOAD + ADAPTER	200	1310	2520	475	1325	200	755
SEPS DRY MASS	1272	1272	1273	1279	1272	1272	1284
LOW THRUST PROPELLANT	616	459	217	221	423	460	1229
TOTAL SEPS/PAYLOAD	2088	3041	4010	1975	3020	1932	3268
LAUNCH VEHICLE ADAPTER	70	100	200	70	100	70	200
TOTAL LAUNCH WEIGHT	2158	3141	4210	2045	3120	2002	3468
LAUNCH VEHICLE CAPABILITY	2820	3420	5080	4100	3270	5800	4400
LAUNCH VEHICLE MARGIN	662	279	870	2055	150	3798	932

*IUS = EXPENDED STRETCHED TRANSTAGE
TUG = BASELINE 30 CRYO TUG
5KS – 5K Wp KICK STAGE
14KS = 14K Wp KICK STAGE
TKS = TANDEM KICK STAGE

Candidate launch vehicles include the Shuttle/Stretched Transtage IUS, the Shuttle/Baseline Cryogenic Tug, or either of these augmented with a 2 270-kilogram (5,000-lbm) kick stage (5KS), a 6 350-kilogram (14,000 lbm) kick stage (14KS), or a combination tandem kick stage (TKS) The required launch vehicle is shown in table 6 1-3 for each mission, the selection criteria were launch vehicle availability, launch energy (C_3), and mass properties Adequate launch vehicle margin exists to comfortably commit to any of the candidate SEPS planetary missions

The primary planetary mission for system design specification, as directed by contract, is the 1981 Encke Rendezvous A representative mission profile and trajectory characteristics are shown in figure 6 1-6

6 1 3 System Functional Requirements

The basic function of the SEPS is to serve as a space transportation system for both planetary and Earth-orbital payloads To accommodate the specified mission model, SEPS must be capable of providing the following functions

1 Primary propulsion for planetary and Earth orbital missions
2 Electrical power to all stage subsystems
3 Control of the stage attitude
4 Capability for storage and implementation of a flight program
5 Capability for data processing
6 Means for transmission of stage engineering data to ground and receipt of commands from ground
7 Stage thermal control
8 Means for determining the stage spatial position
9 Structural support for all stage subsystem elements
10 Payload structural, power, command, and telemetry support
11 Noninterfering stable platform for performing science experiements (attached science packages)
12 Capability to rendezvous, dock, and exchange payloads with and refuel from the IUS and Tug vehicles for Earth-orbital missions
13 Capability to rendezvous and dock with multiple payloads for Earth-orbital missions

The accommodation of system-level functional requirements are allocated to specific stage subsystems These subsystems and their functional interfaces are illustrated in the functional block diagram of figure 6 1-7

6 1 4 External Interfaces

SEPS external interfaces consist of payload, launch vehicle, and the ground data system

SEPS/payload interface requirements are summarized in tables 6 1-4, 6 1-5, and 6 1-6 The specified requirements reflect both payload design factors and operational aspects of mission design Table 6 1-4 presents general (mission independent) requirements while tables 6 1-5 and 6 1-6 address planetary and Earth-orbital missions, respectively

Table 6 1-7 summarizes the SEPS/Launch Vehicle interface requirements Included are requirements derived from operational requirements such as Earth-orbital payload exchange and SEPS refueling Mission applicability is shown and requirement source noted where appropriate

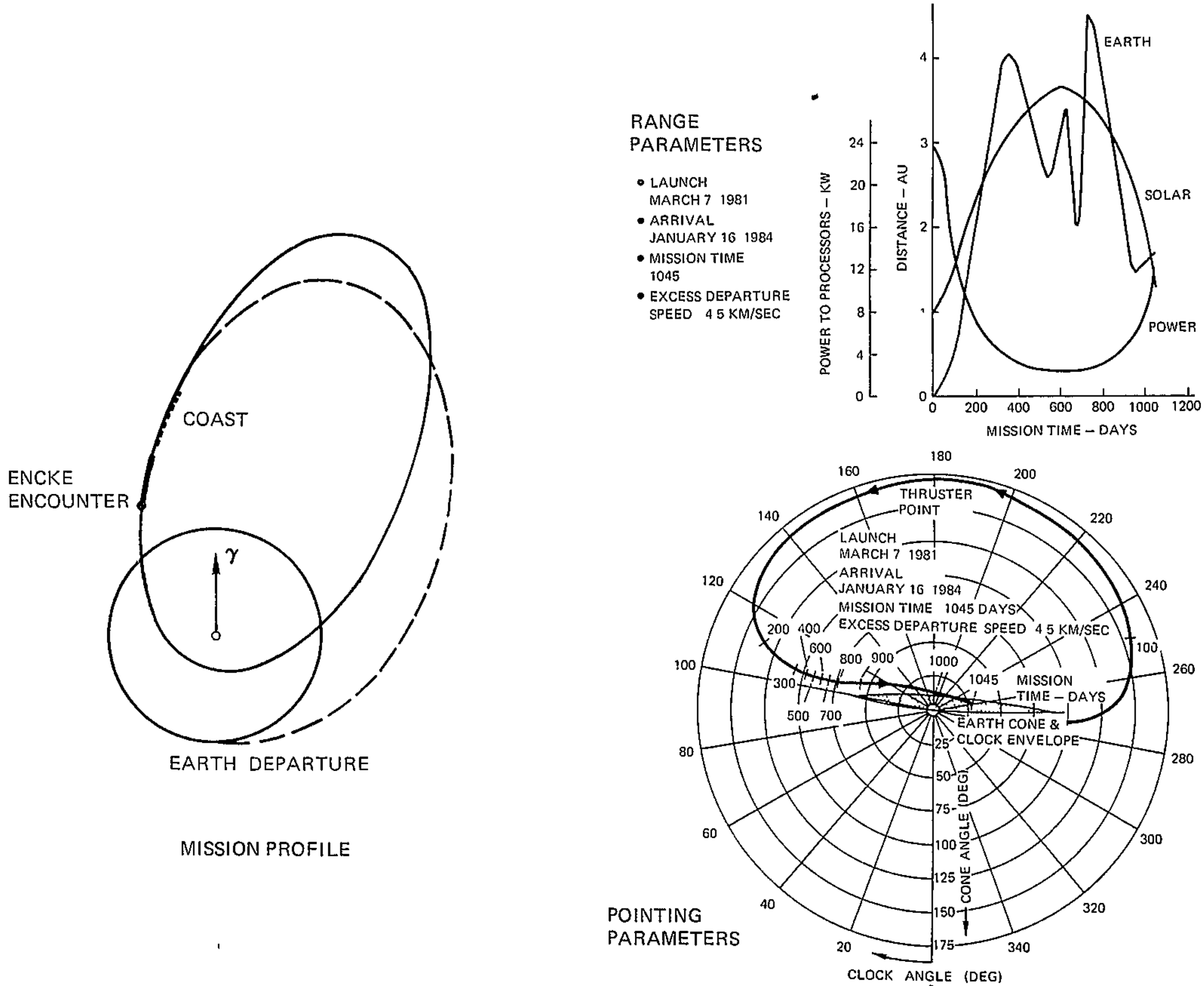


Figure 6 1-6 Encke Rendezvous Mission Characteristics

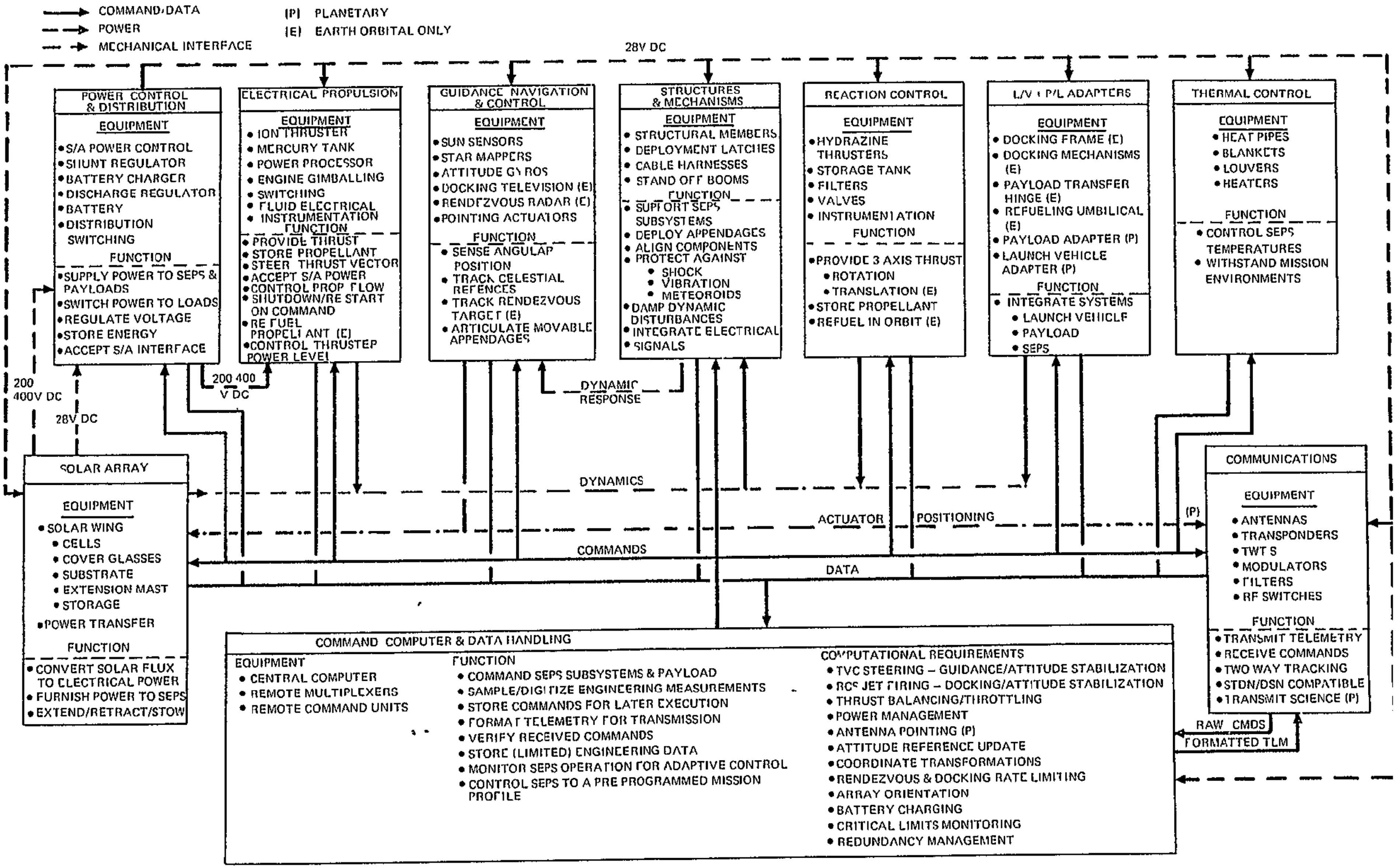


Figure 6 1 7 System Functional Block Diagra

Table 6 1-4 SEPS/Payload Interface Requirements — General

INTERFACE CATEGORY	REQUIREMENT	COMMENTS
I OPERATION	A) SEPS WILL PROVIDE ALL ATTITUDE CONTROL FUNCTIONS FOR SEPARABLE PAYLOAD SPACE CRAFT PRIOR TO SEPARATION	SUBSEQUENT TO LAUNCH VEHICLE SEPARATION
II MECHANICAL	A) SEPS WILL PROVIDE A COMMON PAYLOAD INTERFACE DIFFERING MISSION PAYLOADS WILL BE ACCOMMODATED BY PAYLOAD UNIQUE ADAPTERS	
	B) SEPARATION ΔV TO SEPARABLE PAYLOADS WILL BE SUPPLIED BY SEPS	
III ELECTRICAL	A) SEPS SHALL DISTRIBUTE UMBILICAL AND LAUNCH VEHICLE SUPPLIED POWER TO THE PAYLOAD	
	B) PRE SUN ACQUISITION PAYLOAD POWER SHALL BE SUPPLIED BY THE PAYLOAD	
	C) SEPS PAYLOAD WIRING SHALL INTERFACE AT A SEPARABLE CONNECTOR	
IV ENVIRONMENT	A) THE PAYLOAD DESIGN SHALL ACCOMMODATE THE NATURAL AND LAUNCH INDUCED ENVIRONMENTS	
	B) SEPS DESIGN SHALL ASSURE THAT CONSTRAINTS IMPOSED ON THE PAYLOAD ENVIRONMENT (THERMAL/SOLAR, RADIATION, CONTAMINATION ETC) ARE NOT VIOLATED DUE TO SEPS OPERATION	

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Table 6 1-5 SEPS/Payload Interface Requirements — Planetary Missions

INTERFACE CATEGORY	REQUIREMENT	COMMENTS
I OPERATIONS	A) PAYLOAD SCIENCE DATA ACQUISITION WILL BE LIMITED TO NON THRUSTING PERIODS	
	B) SEPARABLE PAYLOAD SPACECRAFT OPERATION WHILE ATTACHED TO SEPS WILL BE CONSTRAINED BY THE AVAILABLE POWER	EXCEPT SELF POWERED PAYLOADS
	C) SEPS ATTITUDE CONTROL CAPABILITY WILL BE CONSISTENT WITH ATTACHED SCIENCE PACKAGE REQUIREMENTS	PAYLOAD DEPENDENT LIMITS TO BE DETERMINED
II MECHANICAL	A) SEPS SHALL ACCOMMODATE PAYLOAD MASS, INCLUDING ADAPTER, OF UP TO	
	1) 2520 KG FOR SEPARABLE SPACECRAFT	1983 VENUS MAPPER
	2) 200 KG FOR ATTACHED SCIENCE PACKAGE	1981 ENCKE RENDEZVOUS
III ELECTRICAL	A) A MAXIMUM OF 150 WATTS OF SEPS SOLAR ARRAY DERIVED POWER WILL BE PROVIDED FOR PAYLOAD HOUSEKEEPING DURING CRUISE	ALL PAYLOAD TYPES
IV COMMAND & DATA	A) SEPS SHALL PROVIDE A COMMUNICATION LINK FROM PAYLOAD TO GROUND AS FOLLOWS	
	1) 25 KBPS SCIENCE DATA	ATTACHED SCIENCE PACKAGE ONLY @ 1 4 AU (ENCKE RENDEZVOUS ENCOUNTER)
	2) 30 BPS ENGINEERING	ALL PAYLOADS @ 4 6 AU (SATURN ORBITER/PROBE SEPARATION)
	B) SEPS SHALL ISSUE EITHER STORED OR GROUND RECEIVED COMMANDS TO THE PAYLOAD	
	C) COMMAND OR DATA HANDLING ELECTRONICS REQUIRED FOR PROPER SEPS/PAYLOAD INTER FACING SHALL BE CONSIDERED AS PAYLOAD EVEN IF SEPS MOUNTED	

Table 6 1-6 SEPS/Payload Interface Requirements — Earth Orbital Missions

INTERFACE CATEGORY	REQUIREMENT	COMMENTS
I OPERATIONS	A) SEPS SHALL BE CAPABLE OF TRANSPORTING AND DEPLOYING UP TO 4 PAYLOADS ON A SINGLE SORTIE	TRAFFIC ANALYSIS
	B) SEPS SHALL BE CAPABLE OF MULTIPLE PAY LOAD RENDEZVOUS AND DOCKING MANEUVERS (UP TO 4) ON ANY SINGLE SORTIE MISSION	TRAFFIC ANALYSIS DERIVED
	C) SEPS SHALL BE THE ACTIVE VEHICLE IN ALL RENDEZVOUS AND DOCKING MANEUVERS THE TARGET PAYLOADS SHALL BE 3 AXIS STABILIZED WITH SEPS COMPATIBLE DOCKING AIDS AS REQUIRED	
	D) A UNIVERSAL DOCKING SYSTEM SHALL BE USED	
II MECHANICAL	A) THE SEPS/PAYLOAD DOCKING STRUCTURE AND ASSOCIATED MECHANISMS SHALL BE THE SOLE MECHANICAL ATTACHMENT AND AS SUCH SHALL BE CAPABLE OF WITHSTANDING THE LAUNCH ENVIRONMENT AS APPROPRIATE	
	B) SEPS SHALL ACCOMMODATE A TOTAL PAYLOAD MASS OF UP TO 4500 KG	TRAFFIC ANALYSIS
III ELECTRICAL	A) SEPS SHALL PROVIDE ELECTRICAL POWER TO PAYLOAD(S) AS FOLLOWS	
	1) THRUST PHASES 500 WATTS TOTAL 28V DC THERMAL CONTROL POWER DISTRIBUTED TO SORTIE PAYLOADS AS REQUIRED	MAX AVAILABLE POWER WITH 25 KW SOLAR ARRAY
	2) SUN OCCULTATION 400 WATTS TOTAL 28V DC THERMAL CONTROL POWER DISTRIBUTED TO SORTIE PAYLOADS AS REQUIRED	MAX AVAILABLE WITH BATTERY SIZED FOR DOCKING MANEUVERS AT 50% DEPTH OF DISCHARGE
	3) PAYLOAD PRE DEPLOYMENT CHECKOUT 500 WATTS MAXIMUM 28V DC POWER FOR PAYLOAD ENGINEERING CHECKOUT	THERMAL CONTROL OF OTHER ATTACHED PAYLOADS MAY REDUCE AVAILABLE POWER
	B) ELECTRICAL CONNECTORS SHALL ACCOMMODATE MULTIPLE DOCKING AND RELEASE WITH NO PREDICTABLE END OF LIFE	
	C) PAYLOAD THERMAL CONTROL HEATER CIRCUITS SHALL BE INDEPENDENT WITH POWER CONTROL LED BY SEPS	PAYLOAD SUBSYSTEMS ARE UNPOWERED THUS PROVIDING NO SWITCHING CAPABILITY
IV COMMAND & DATA	A) SEPS SHALL ISSUE EITHER STORED OR GROUND RECEIVED COMMANDS TO ANY PAYLOAD	
	B) SEPS SHALL PROVIDE A COMMUNICATIONS LINK FROM ANY PAYLOAD TO THE GROUND AT A RATE NOT TO EXCEED 30 BPS	

Table 6 1-7 SEPS/Launch Vehicle Interface Requirements

INTERFACE CATEGORY	REQUIREMENT	MISSION APPLICABILITY		COMMENTS
		EO	PLAN	
I GENERAL	A) SEPS DESIGN MUST BE COMPATIBLE WITH THE FOLLOWING LAUNCH VEHICLES			CONSTRAINT (MSFC)
	1) SHUTTLE/IUS	x	x	
	2) SHUTTLE/TUG	x	x	
	3) SHUTTLE/IUS (TUG)/KICK STAGE		x	
	B) SEPS AND IUS (TUG) MUST BE CAPABLE OF ORBITAL RENDEZVOUS AND DOCKING	x		
	C) PROVISIONS SHALL BE MADE FOR ON ORBIT EXCHANGE OF PAYLOADS BETWEEN SEPS AND TUG IN ORBIT DESIGN SHALL NOT PRECLUDE ONE WAY PAYLOAD TRANSFER ONLY	x		UP AND DOWN PAYLOADS ACCOMMODATED WITH RECOVERABLE TUG
	D) PROVISIONS SHALL BE MADE FOR ON ORBIT PAYLOAD TRANSFER FROM THE IUS TO SEPS	x		IUS EXPENDABLE – NO DOWN PAYLOADS
	E) PROVISIONS SHALL BE MADE FOR SEPS ON ORBIT REFUELING VIA IUS OR TUG SERVICING SHALL BE LIMITED TO PROPULSION AND ATTITUDE CONTROL PROPELLANTS/PRESSURANTS ONLY	x		SEPS FUELED FOR SINGLE SORTIE ONLY TO MINIMIZE WEIGHT/OPTIMIZE PERFORM ANCE
	F) ALL SEPS/LAUNCH VEHICLE MECHANICAL AND ELECTRICAL PHYSICAL INTERFACES SHALL ACCOMMODATE MULTIPLE OPERATIONS	x		EARTH ORBITAL OPERATIONS INCLUDING SEPS RECOVERY
II MECHANICAL	A) SEPS SHALL PROVIDE A COMMON STRUCTURAL INTERFACE FOR ALL LAUNCH VEHICLES VARYING MISSION/LAUNCH VEHICLE DEPEN DENT INTERFACES SHALL BE ACCOMMODATED BY UNIQUE ADAPTERS	x	x	DESIGN GOAL
	B) LAUNCH VEHICLE MECHANICAL INTERFACES SHALL BE IN ACCORDANCE WITH THE FOLLOWING DOCUMENTATION			
	1) SHUTTLE 'SHUTTLE BASELINE ACCOMMODATION FOR PAYLOADS," JSC 07700 VOL XIV	x	x	
	2) TUG 'BASELINE SPACE TUG ' MSFC 68 M00039, VOLS 1 4	x	x	
	3) IUS TBD	x	x	
	4) KICK STAGE (S) TBD		x	
III ELECTRICAL	A) THE SHUTTLE/TUG/IUS SHALL PROVIDE PRE SEPARATION POWER TO BOTH THE SEPS AND ITS ATTACHED PAYLOAD(S)	x	x	LAUNCH AND PAYLOAD EXCHANGE OPERATIONS
	B) ELECTRICAL INTERFACES SHALL BE IN ACCORDANCE WITH DOCUMENTATION NOTED IN II(b)	x	x	
IV ENVIRONMENT	A) SEPS/PAYLOAD DESIGN SHALL ACCOMMODATE PRELAUNCH/LAUNCH ENVIRONMENTS SPECIFIED IN DOCUMENTATION NOTED IN II(b)	x	x	

6.1 5 Mass Properties

Table 6 1-8 summarizes basic SEPS weight for both the planetary and Earth-orbital configurations All SEPS weight elements are accounted for except mercury propellant which is mission dependent Representative maximum propellant loading is noted for each class of mission

A weight growth allocation of 12 3 percent has been included in the total SEPS dry weight This value represents the upper bound of expected weight growth as shown in table 6.1-9 Basis for the weight growth shown for each subsystem is the result of a detailed analysis considering hardware and technology design status and overall SEPS system design phase The estimated "probability of not exceeding" this weight growth is 50 percent

6.1.6 Reliability

The results of SEPS reliability analyses are summarized in table 6 1-10 Two planetary missions were examined, the Encke Rendezvous mission, which represents the longest total burn time (1,005 days, one to eight thrusters operating) and the Mercury Orbiter which requires the longest single thruster burn duration (315 days, eight thruster continuous operation) A representative Earth-orbital mission was examined on the basis of a single sortie, the combination of nine sorties, and finally, the accumulation of nine sorties plus intervening coast time In all cases, both the number of thrusters and PPU's installed and the number required to be operating was the dominant factor in overall system reliability For this reason, various combinations were examined with the results as shown The target stage reliability is 90 for a single launch Encke Rendezvous mission To achieve this target with the capability to operate 8 thrusters simultaneously, as required in the mission design, the installation of 10 thrusters and 9 PPU's (total) is required This configuration results in stage total of 881 for the Mercury Orbiter and 968 for the single sortie Earth-orbital mission The stage reliability drops off, as noted, when the number of sorties and coast time accumulate

6 2 SUBSYSTEM ANALYSIS

During this study, subsystem designs have been postulated that satisfy all applicable mission and system requirements The design of all SEPS subsystems is based on technology expected to be available in the 1976-1977 time period The major components of the vehicle are summarized in table 6 2-1 The majority of this hardware is either off the shelf or may be designed with available (present-day) technology The items that require further development are the ion thrusters, the power processor units, the mercury tank, and the solar array A continuation of the development in progress through the SEP-AST program, combined with the SRT programs recommended by this study, should result in the availability of these components by 1977

The dynamic characteristics of the solar arrays (from previous studies) have been integrated into a total vehicle dynamic model for the 1038-7 design This model has been exercised through a range of configuration variations to generate the modal response characteristics for the stage As shown in table 6 2-2, the parameters that were varied for this study are payload weight, mercury propellant slosh frequency, percent of total solar-array deployment, solar-array aspect ratio, and solar-array orientation with respect to the bus Figure 6 2-1 shows a typical output from the analysis for case 5 of the table The modal characteristics have been used as inputs for the SEPS stability model This model, shown in figure 6 2-2, was constructed to simulate the stage behavior whether stabilized by the hydrazine reaction control system, or by thrust vector gimballing This

Table 6 1-8 SEPS Weight Summary

	Weight, Kg	
	Planetary configuration 1038-7/Plan	Earth Orbital configuration 1038-7
Structures/Mechanisms/Cabling	184 1	182 1
Electric Propulsion	360 0	360 0
Communications	29 6	17 4
Command Computer and Data Handling	9 2	9 2
Guidance, Navigation and Control	46 4	80 4
Reaction Control	18 9	18 9
Solar Array	381 6	381 6
Power Control and Distribution	54 0	54 0
Thermal Control	33 5	33 5
Payload and Launch Vehicle Adapters	15 0	54 6
Allocated weight growth	139 3	146 5
Total dry weight	1,271 6	1,338 2
RCS Hydrazine	32 6	20 8
RCS Nitrogen	7 5	7 4
Mercury Propellant	(1)	(2)
Total SEPS weight excluding mercury propellant	1,311 7	1,366 4

(1) Mission dependent - - Up to 1229 KG per current mission analysis

(2) Sortie dependent - - Up to 404 KG per current traffic analysis

Table 6 1-9 SEPS Expected Weight Growth (Earth Orbital)

SUBSYSTEM	BASIC IDENTIFIED WEIGHT (KG)	WEIGHT GROWTH (KG)		BASIS FOR WEIGHT GROWTH (KG)
		MINIMAL	TYPICAL	
03-01 STRUCTURE MECHANISMS/ CABLING	182 1	18 2	36 4	10% TO 20% – MVM '73 STRUCTURES AND MECHANISMS GROWTH WAS 19 3% AND CABLING WAS 29 4%
03 02 ELECTRICAL PROPULSION	360 0	19 2	44 4	+0 5 TO 1 0 KG ON EACH THRUSTER, +1 0 TO 3 0 KG ON EACH PPU,5% TO 10% ON GIMBALLING, PROPULSION CABLING AND LOW THRUST PROPELLANT SUBSYSTEM
03 03 COMMUNICATIONS	17 4	0 9	1 7	5% TO 10% – SUBSYSTEM IS FAIRLY WELL DEFINED WITH USE OF MVM '73 OR VO '75 COMPONENTS
03 04 COMMAND COMPUTER & DATA HANDLING	9 2	0 9	3 8	10% TO 40% – DATA HANDLING REQUIREMENTS NOT WELL DEFINED AT THIS TIME – MINIMAL SYSTEM IS CURRENTLY IDENTIFIED
03 05 GUIDANCE, NAVIGATION & CONTROL	80 4	2 4	4 8	3% TO 6% – COMPONENTS FAIRLY WELL DEFINED AND WEIGHTS SUPPORTED BY VENDOR DATA
03 06 REACTION CONTROL SYSTEM	18 9	1 0	1 9	5% TO 10% – HYDRAZINE TANK IS MAIN WEIGHT VARIABLE – OTHER COMPONENTS MOSTLY OFF-THE-SHELF
03 07 SOLAR ARRAY ASSEMBLY	381 6	15 3	30 5	4% TO 8% – LMSC SUGGESTED 9 4% (D384232) HOWEVER RECENT CHANGES AND EXPECTED CELL EFFICIENCY IMPROVEMENT INDICATE LOWER GROWTH
03 08 POWER CONTROL & DISTRIBUTION	54 0	2 7	5 4	5% TO 10% – BATTERY REQUIREMENTS FAIRLY WELL DEFINED
03 09 THERMAL CONTROL	33 5	3 4	6 7	10% TO 20% – BASIC IDENTIFIED WEIGHT IS BASED UPON MINIMUM WEIGHT HEAT PIPE SYSTEM
03-10 PAYLOAD AND LAUNCH ADAPTERS	54 6	8 2	10 9	15% TO 20% – DOCKING AND PAYLOAD TRANSFER NOT WELL DEFINED
TOTAL SEPS DRY WEIGHT	1191 7	72 2	146 5	THIS REPRESENTS 6 0% TO 12 3% OF THE TOTAL DRY WEIGHT

Table 6 1-10 Mission Reliability Summary

Mission \ Subsystem	Struc	Comm	Comp & Data	RCS	Therm	Guid & Nav	Solar Array [1]	Power Distribution	Stage less EPS	EPS	Thrusters/PPU's Installed	Thrusters/PPU's Minimum operating	Stage total	Comments
Planetary														
Enke Rendezvous	0 9999	0 9718	0 9773	0 9919	0 9999	–	0 9986	0 9822	0 920	0 9883	10/9	8/8	0 908	Baseline
										0 9700	10/8	8/8	0 891	
										0 9925	10/9	7/7	0 912	[2]
										0 9920	10/8	7/7	0 912	
Mercury Orbiter (7560 hours)	0 9999	0 9978	0 9914	0 9938	0 9999	–	0 9997	0 9997	0 982	0 8968	10/9	8/8	0 881	Baseline
										0 7266	10/8	8/8	0 714	
										0 9773	10/9	7/7	0 959	[2]
										0 9565	10/8	7/7	0 939	
Earth Orbit														Baseline single longest sortie 3360 hours
Single sortie	0 9999	0 9990	0 9961	0 9923	0 9999	0 9982	0 9997	0 9998	0 985	0 9830	10/9	8/8	0 968	
Product 9 sorties	0 9999	0 9810	0 9766	0 9632	0 9998	0 9940	0 9956	0 9969	0 910	0 9323	10/9	8/8	0 848	[3]
Cumulative 9 sorties plus coast time	0 9999	0 9510	0 961	0 8620	0 9996	0 9928	0 9952	0 9940	0 773	0 4210	10/9	8/8	0 319	[4]
										0 2560	10/8	8/8	0 198	
										0 7630	10/9	7/7	0 591	[2]
										0 9350	10/9	6/6	0 724	

Notes

[1] Lockheed reports 50 cycle, 5 year reliability of 989

[2] Baseline flight time is increased with fewer thrusters operating

[3] An upper bound reliability $\prod_{i=1}^{9} R_i$ does not allow for sortie to sortie failures

[4] A lower bound reliability represents 4 year life, including failures during thrust and coast periods

Table 6 2-1 SEPS Subsystem Technology Status

Subsystem	Major Component	Technology Status
Structures & Mechanisms	—	Available
Electrical Propulsion	Ion thruster	AST + SRT (1977)
	Power processor	AST (1976)
	Mercury storage	SRT (1976)
	Switch matrix	Available
Communications	Antenna	Available
	Transponder	Off-shelf
	TWT	Off-shelf
	Modulator	Off-shelf
Command, Computer & Data Handling	Computer	Available
	Remote multiplexer	Off-shelf
	Remote command unit	Off-shelf
Guidance, Navigation & Control	Gyro	Off-shelf
	Star mapper	Off-shelf
	Sun sensor	Off-shelf
	Docking television	Available
	Rendezvous radar	Available
	Pointing actuators	Off-shelf
Reaction Control	Hydrazine thrusters	Off shelf
	Tankage	Available
Solar Array	Solar cells	Available
	Flexible array	AST + SRT (1977)
	Power transfer	Off shelf
Power Control & Distribution	Shunt regulator	Off shelf
	Battery	Off-shelf
	Battery charger	Off-shelf
	Battery discharge regulator	Off-shelf
	Solar panel controller	Available
	Power distribution	Available
Thermal Control	Heat pipes	Off shelf
	Louvers	Off-shelf
	Blankets	Available
Adapters	Docking frame	Available
	Refueling umbilical	Available

Table 6 2-2 Dynamic Characteristics Study SEPS-7 Configuration Parameter Variations

CASE NO	NO OF P L S	SOLAR ARRAY	% DEPLOY	PANEL TILT ANGLE (DEG)	SLOSH FREQ (HZ)
1	0	9 39	100	0	–
2	1	9 39	100	0	–
3	2	9 39	100	0	–
4	3	9 39	100	0	–
5	4	9 39	100	0	–
6	0	9 39	50	0	–
7	0	9 39	25	0	
8	4	6 3	100	0	–
9	4	15 5	100	0	–
10	4	9 39	100	0	0 5
11	4	9 39	100	0	2 0
12	0	9 39	100	0	2 0
13	4	9 39	100	0	5 0
14	4	9 39	100	30	
15	4	9 39	100	45	–
16	4	9 39	100	90	–
17	0	9 39	0	0	–

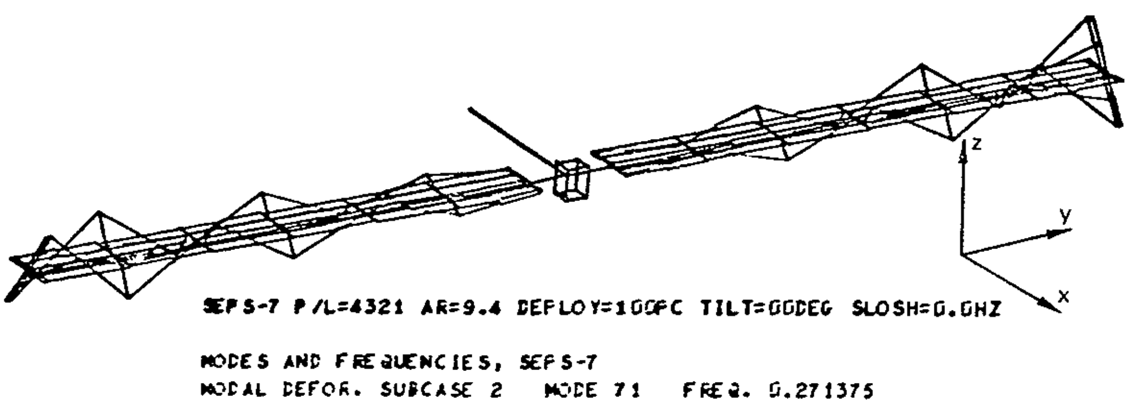


Figure 6.2-1. 1st Mast Torsion

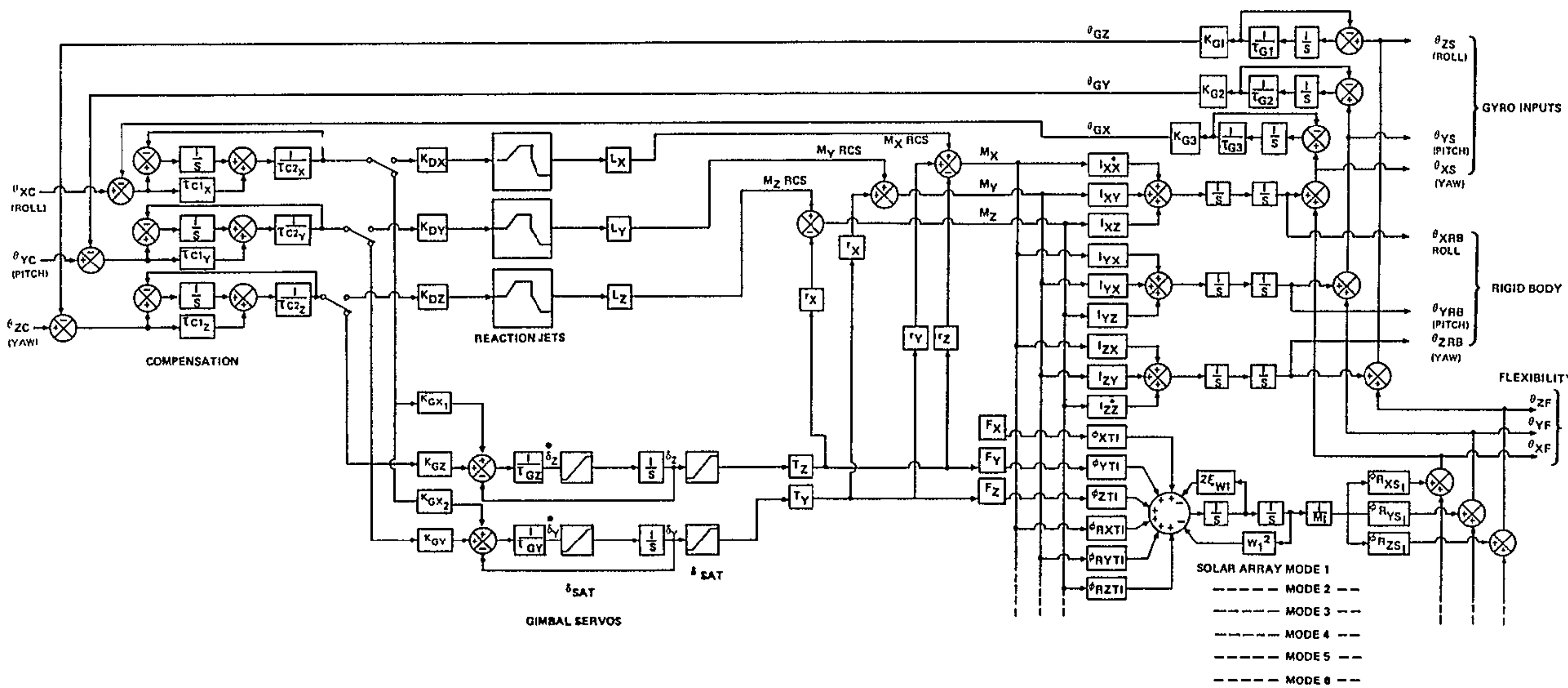


Figure 6 2 2 SEPS Stability Model

investigation is not yet complete, but, at this time, no major SEPS stability problems have been found nor are any anticipated Mercury slosh is probably not as big a problem as earlier studies had indicated The solar-array dynamics dominate the vehicle transfer functions due to their larger mass and lower frequency The concept of refueling the SEPS for each Earth-orbital sortie has been adopted, and this decreases the likelihood of a problem from Mercury slosh since both the mass of propellant onboard and the size of the tankage are reduced Parametric studies of SEPS dynamics and control system stability should be completed to ensure that no undesirable interactions exist In particular, cross-axis coupling due to solar-array tilt will require further investigation and allowable payload center-of-gravity envelopes and dynamics must be defined

An analysis of the requirements for flight performance reserves for an electrically propelled vehicle has revealed that this factor cannot be treated in the same manner as for an impulsive rocket The possibilities of thruster performance variations must be factored into the design of a mission trajectory to provide an opportunity to utilize the flight performance reserves As pointed out in table 6 2-3, the outer planet missions and the rendezvous missions (as designed in all feasibility studies to date) contain no provisions for additional thrust periods, which are necessary to use any reserve fuel that might be carried These missions should be designed to the minimum probable thruster performance This requires a much more detailed knowledge of thruster performance statistics than is presently available, in order to avoid unreasonably conservative mission design It is seen in figure 6 2-3 that with thruster performance variations in the range of 1 to 3 percent (as is presently understood), the electrical propulsion efficiency must be reduced by 5 to 15 percent (from the 65 percent shown in table 6 2-4) This requirement for thruster statistical data is also evident from the guidance and navigation error analysis A low-autonomy strategy (one which relies on ground tracking rather than onboard hardware) is used to accomplish the SEPS guidance However, the analysis (fig 6 2-4) indicates that with the present thruster statistics, SEPS requires ground tracking periods at least daily to maintain acceptable adherence to a preplanned trajectory This could be an unacceptable demand on the schedules of the tracking networks

Data from detailed thermal analyses of the ion thrusters, the power processors, the solar array, and other elements of the SEPS configuration have been integrated into a total vehicle thermal model The results from the computer implementation of this model do not indicate any serious problems in accomplishing the stage thermal design Thruster heating during solar occultations will be necessary to avoid performance penalties due to startup delays in Earth-orbital operations Since engine shutdowns (due to eclipse) are possible on a large percentage of the revolutions for these missions, a lengthy startup procedures could cause a dramatic reduction in payload delivery capability Current procedures for AST thrusters are quite leisurely (70 to 90 minutes) As shown in figure 6 2-5, this would result in a SEPS performance loss of 10 to 30 percent when compared to the case of continuous thrusting Thermal transients for a 30°C (ambient) startup are as shown in figure 6 2-6 The significant feature of these data is that the thermal lags are all approximately 10 minutes, except for the isolator flange, which is 40 to 50 minutes By heating this element during eclipse periods, a minimum start time as low as 10 minutes may be feasible for the present thruster design A start lag of 15 minutes appears easily attainable for Earth-orbital operations with a nominal (10 watt/thruster) battery-powered preheat This is recommended for future mission analysis

Reaction control jets were sized to minimize hydrazine usage during limit cycle operations This criteria resulted in a choice of 2 2-newton (0 5-lb_f) thrusters on all axis Common thrusters are

Table 6 2-3 Mission Dependence of FPR

TYPE	CHARACTERISTICS
1 DEEP SPACE	• ZERO POWER AT END OF BURN
• JUPITER ORBITER	∴ DESIGN TO MIN PERFORMANCE OR PROVIDE PAYLOAD ΔV
• SATURN PROBE	
2 FLYBY	• SOFT OBJECTIVE
• ENKE	∴ DESIGN TO NOMINAL PERFORMANCE
• ASTEROID	
• OUT-OF-ECLIPTIC	
3 RENDEZVOUS	• POWER LIMITED
• OUTER	∴ ADD FPR AND NOMINAL COAST
• MARS	OR DESIGN TO MIN PERFORMANCE
• ASTEROID	OR UTILIZE COURSE CORRECTION ROCKET
• COMET	
• INNER	• SURPLUS POWER
• VENUS ORBITER	∴ UPRATE PPU AND ADD FPR OR AS ABOVE
• MERCURY ORBITER	

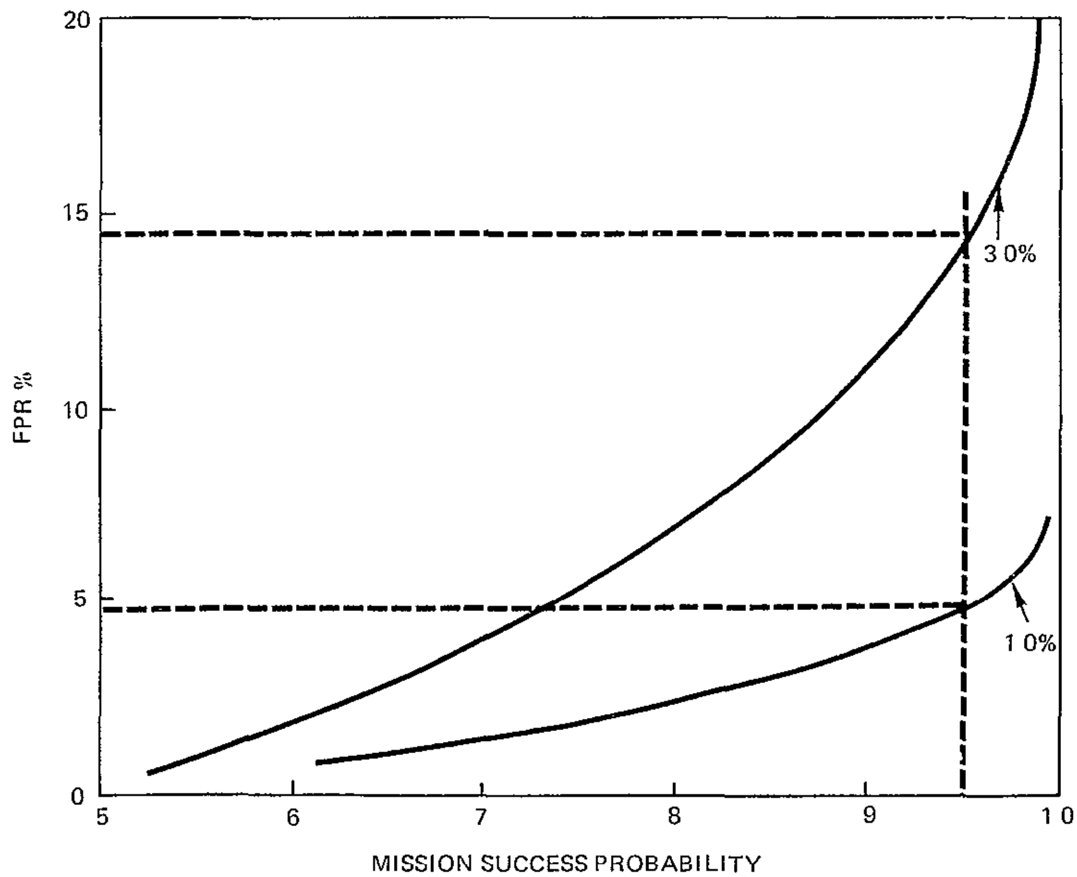


Figure 6 2-3 Flight Performance Reserves Requirement

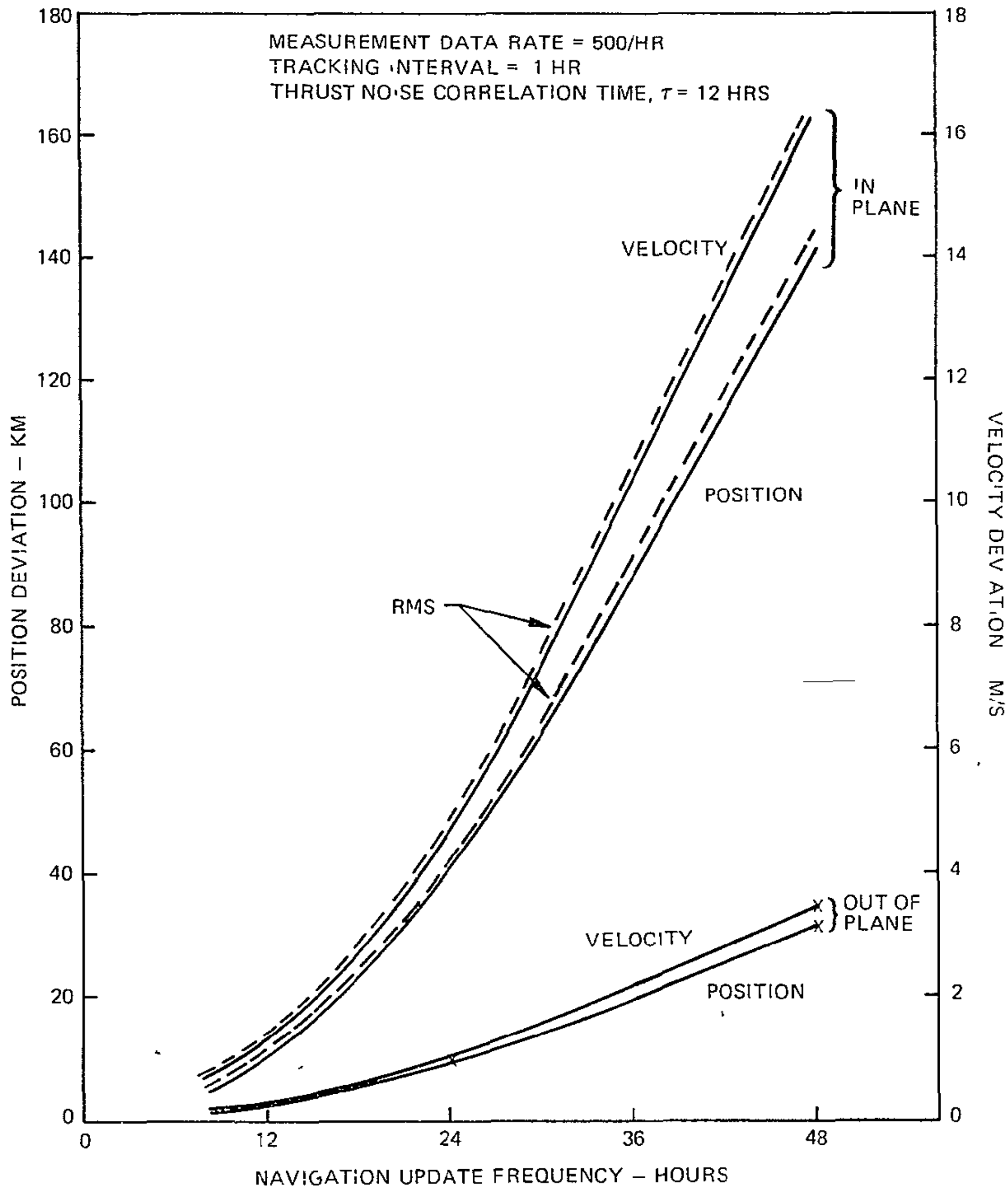


Figure 6 2-4 Trajectory Deviations From Nominal After 6 Days in Orbit

Table 6 2-4 *Baseline EPS Performance Selection*

ASSEMBLY	EFFICIENCY*
THRUSTER	0 718
CABLING	0 999
POWER PROCESSOR	0 9**
EPS OVERALL	0 646

*EXTRAPOLATED TO 2 1A

**MSFC GROUNDRULE

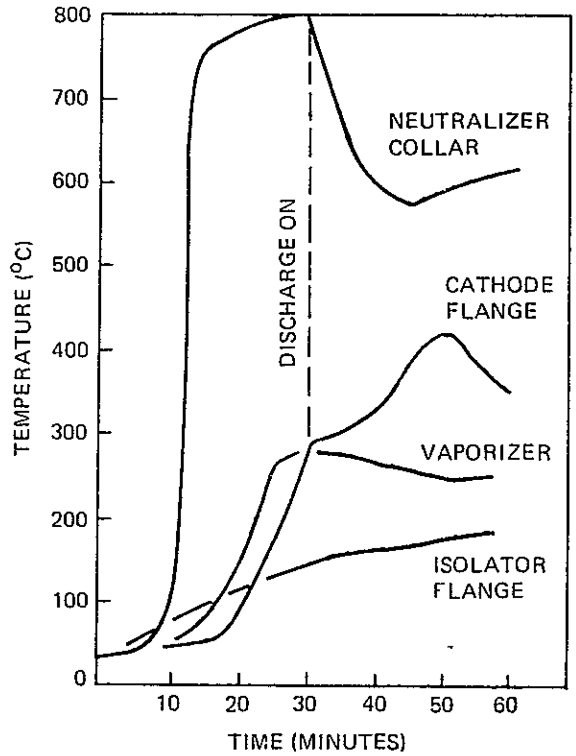


Figure 6 2-6 Ion Thruster Start-Up Thermal Transients

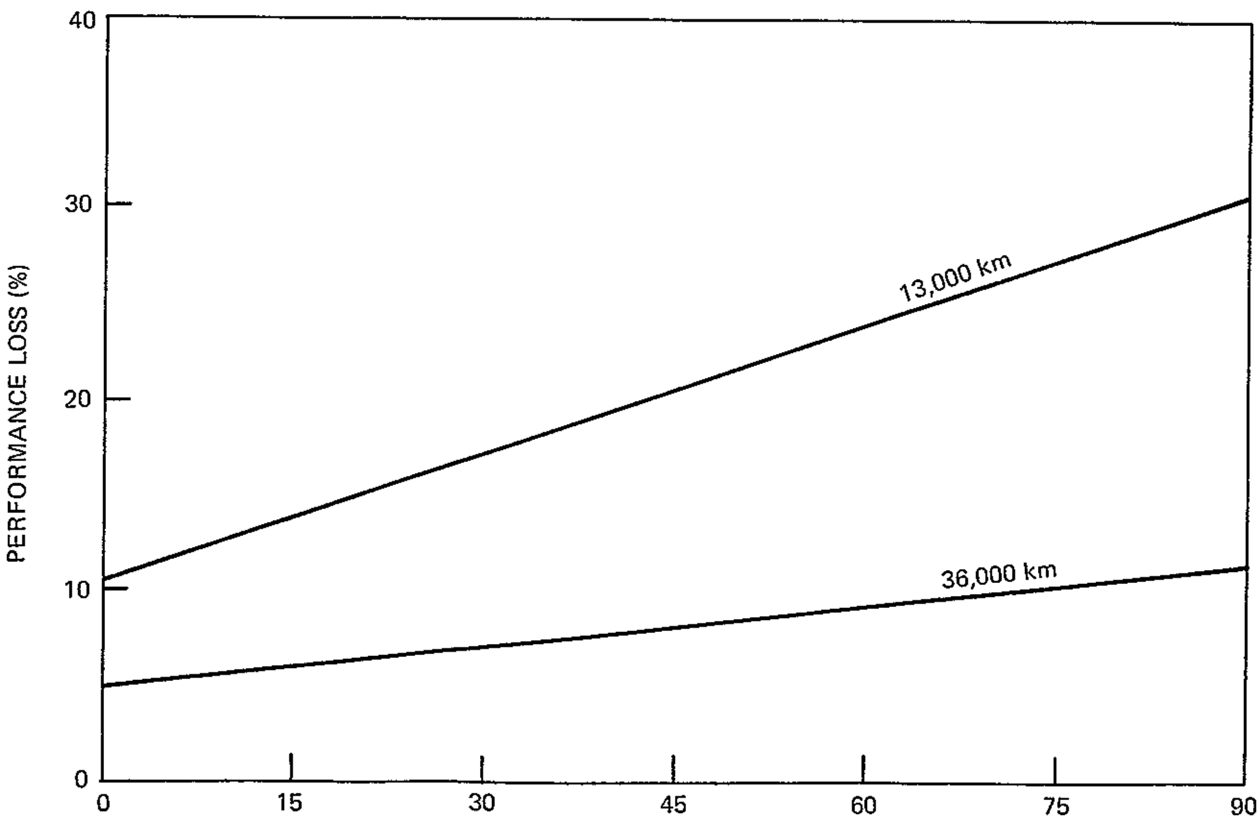


Figure 6 2-5 Startup Effects — Earth Orbital recommended for limit cycle (orientation) control and for translational forces (required for docking) to increase stage reliability and minimize weight For the large payloads anticipated for GEOSEPS, these thrusters result in accelerations of 1 2 mm/sec^2 A gimballed television camera has been included in the design for remote piloting of the docking maneuvers Examination of flight simulator data for remotely piloted maneuvers (see figure 6 2-7) shows that the SEPS accelerations are below the pilot's visual rate-sensing threshold A collision between the SEPS and its docking target (payload or Tug) is thus possible To eliminate this potential hazard, a rate-limiting system will be required for rendezvous and docking A radar system must be included in the system design to supply range and range-rate data With such data, a rate-limiting characteristic (similar to that shown in figure 6 2-8) can be implemented by a simple modification to the RCS control-loop algorithms in the stage computer

The direct energy transfer system is recommended for the control and distribution of SEPS electrical power to maximize subsystem reliability and efficiency As shown in figure 6 2-9, with this concept, the electrical propulsion subsystem is supplied from a high-voltage (200-400 volt) section of the solar array, while the housekeeping subsystems are fed directly from a separate 30-volt section Regulation hardware is kept to a minimum, consisting primarily of those components necessary for the storage of energy for use by the SEPS during eclipse periods Flexible cabling is recommended for the interface between the solar array and the stage The capability of independently rotating the two solar wings will be required to avoid shutting down the electrical propulsion (with the performance loss which that implies) when periodically unwinding the power transfer cabling

Radiation effects will be the major contributor to power output degradation of the SEPS solar array Due to the presence of the Van Allen belts, Earth-orbital missions are more hazardous than planetary ones in this respect

Figure 6 2-10 shows the array degradation of a SEPS spiraling out to synchronous orbit as a function of the initial altitude Constant thrust operation (equivalent to eight thrusters operating at 2 0 amperes) is assumed It is obvious that for transfer orbits below 4 000 kilometers, half of the array will be effectively "thrown away" after one use Thus, it is recommended that the SEPS not be used from low Earth orbits (e g , directly from the Shuttle)

This single pass data can be extrapolated to give the degradation in array output for multiple transportation sorties by the stage (see fig 6 2-11) Here it is seen that the inclusion of a solar flare model results in unacceptable degradation for even single trip usage of the stage The array cannot be exposed to any major solar event It is concluded that the solar wings must be retracted and stored in their containers whenever a major flare is sensed Sensing may be either Earth-based or onboard

When it is assumed that the cells are protected during flares, figure 6 2-12 results The initial array performance has been improved, but the cumulative effects of multiple trips from the lower separation altitudes are still severe Operation from 13 000 kilometers (7,000 nmi) results in an array degradation of 32 percent (8 kilowatts) after five sorties This implies that only five-thruster operation could be sustained during the fifth sortie Alternatively, the array could be designed for an initial output of 36 kilowatts, if full eight-thruster operation is mandatory for five round trips

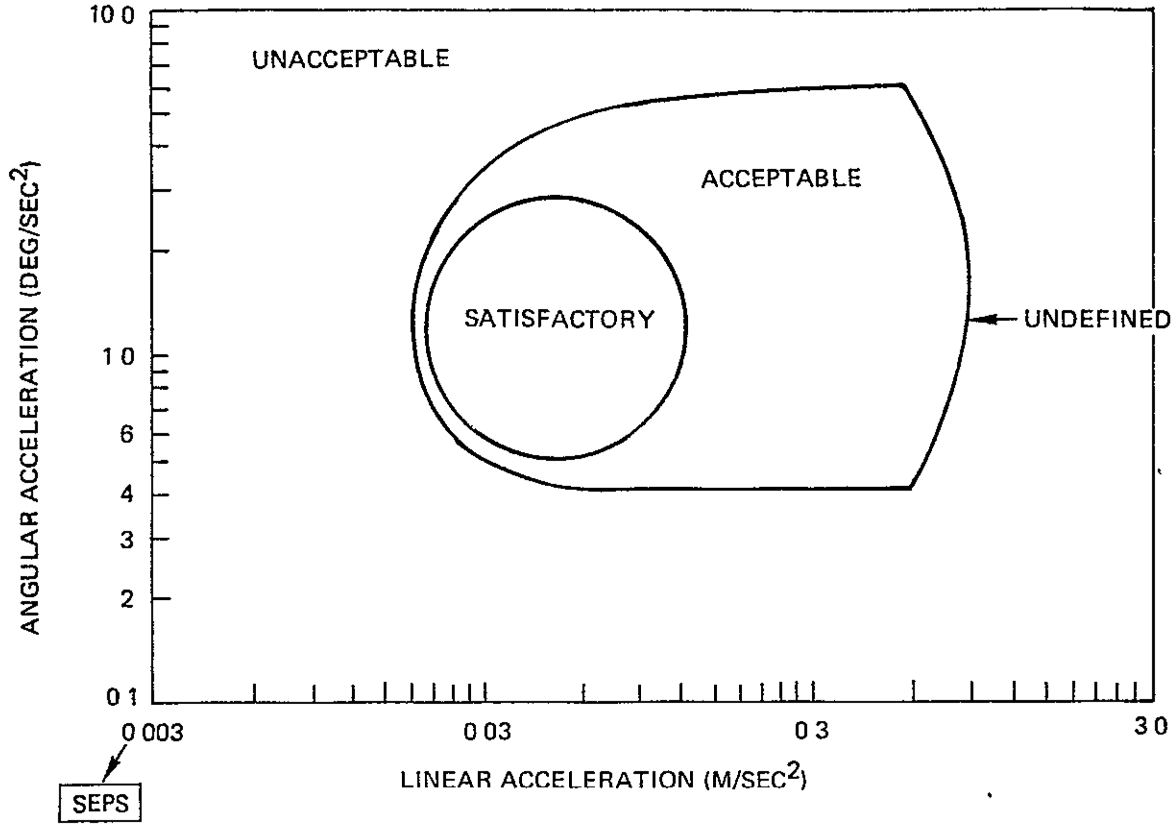


Figure 6 2-7 Pilot Handling, Docking Maneuvers

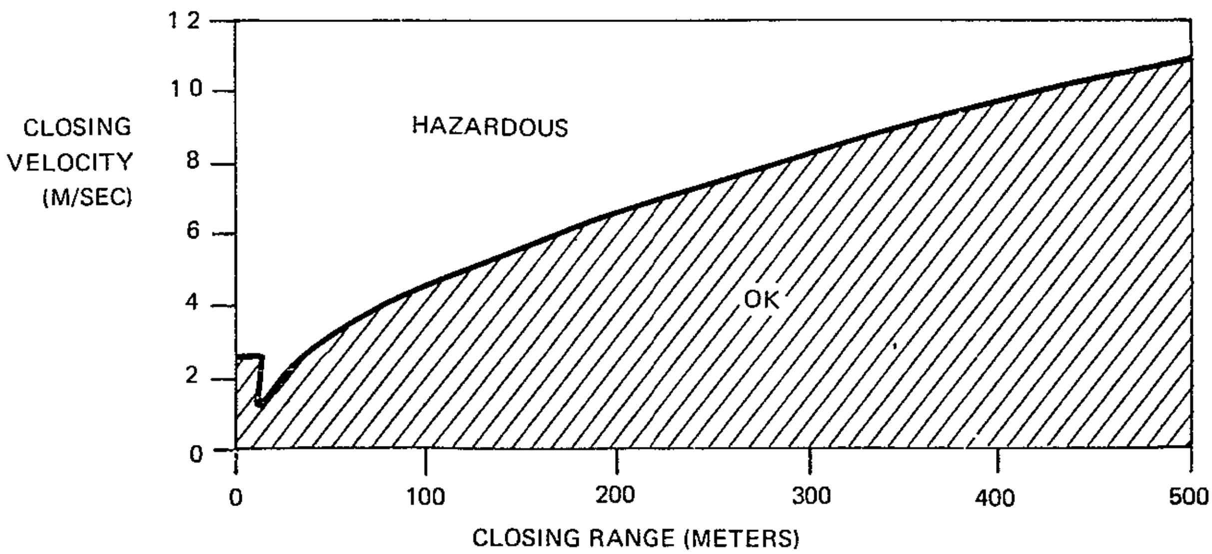


Figure 6 2-8 Range/Range-Rate Limits for Docking

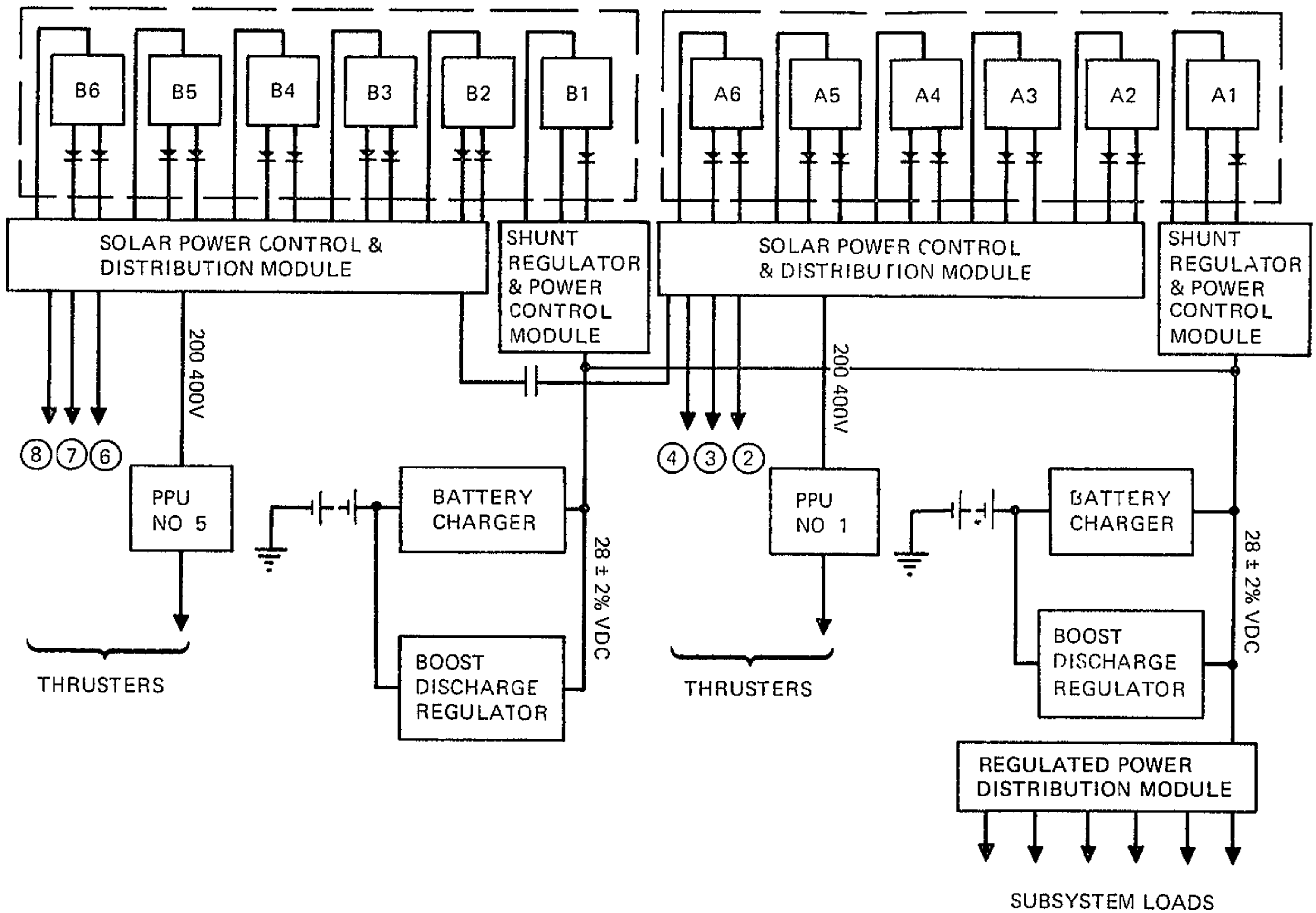


Figure 6 2-9 Direct Energy Transfer System

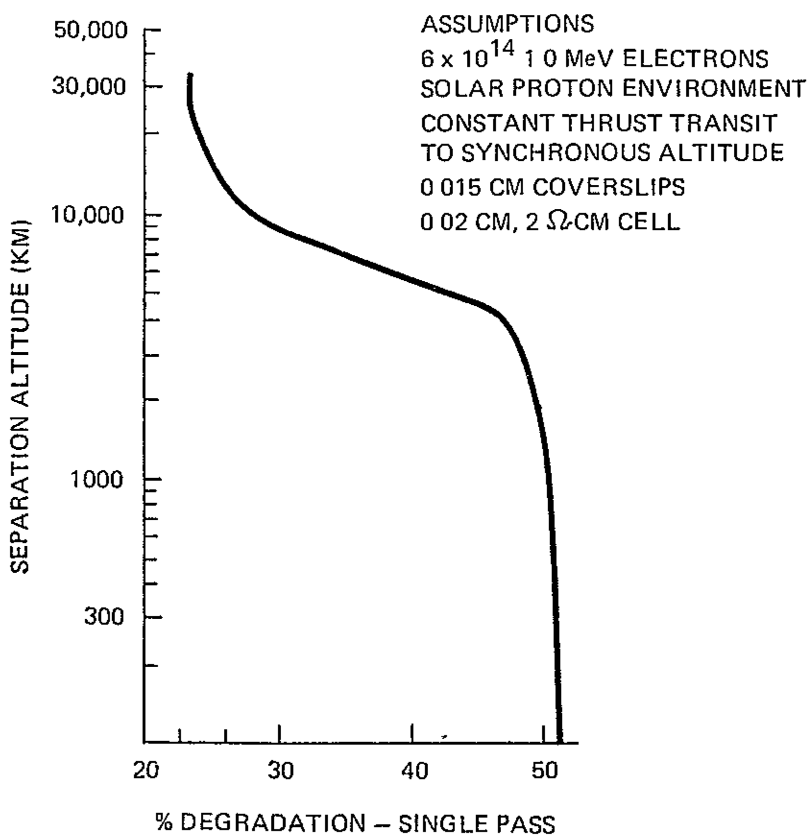


Figure 6 2-10 S/A Degradation With Initial Solar Flare

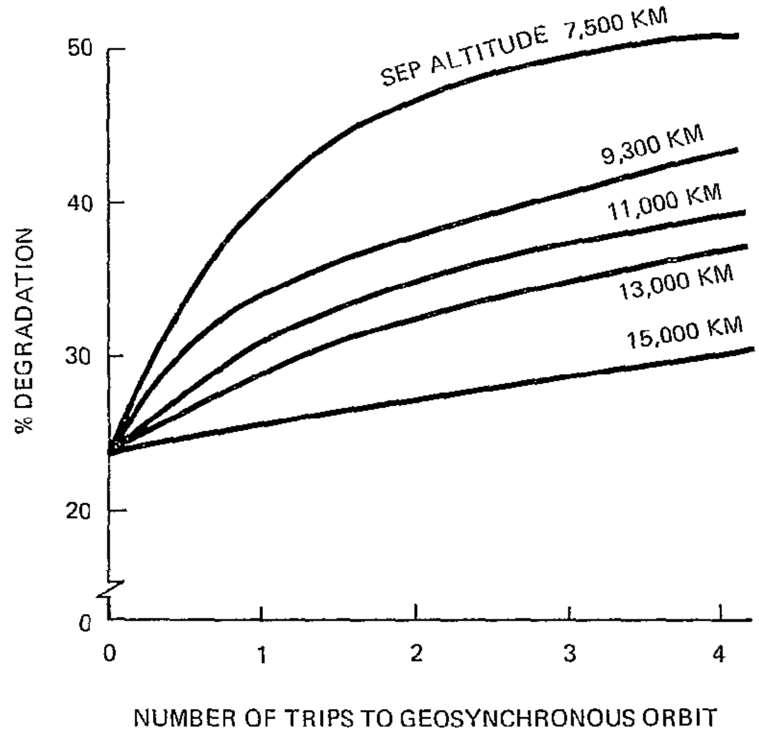


Figure 6 2-11 Effects of S/A Degradation With Initial Solar Flare

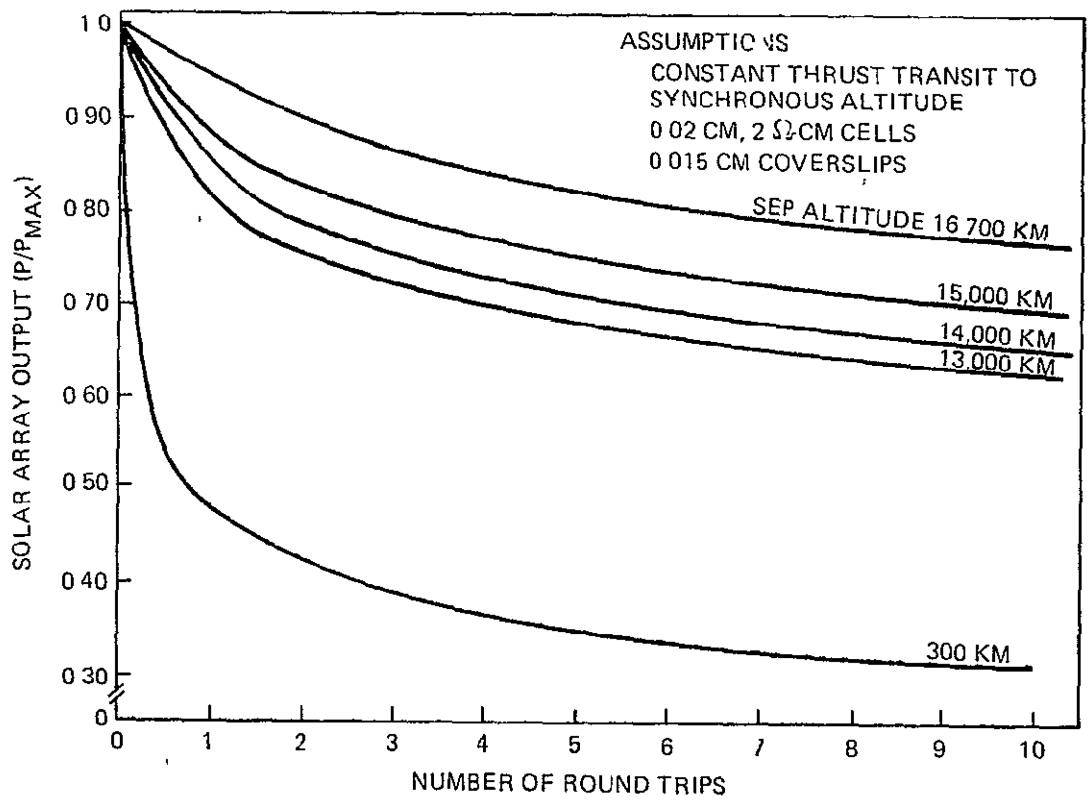


Figure 6 2-12 S/A Degradation with Conventional Cells

Similar data was developed for an array designed with lithium doped, "radiation resistant" cells (see fig 6 2-13) The degradation has been further reduced, but is still serious Evidently then, the key to avoiding unreasonable degradation of the solar array is to increase the altitude of the SEPS/Tug transfer orbit An increase to a 17 000-kilometer (9,000 nautical miles) altitude, when combined with the recommended lithium cells, results in less than 10 percent power degradation over the SEPS lifetime The Boeing traffic analysis (see section 6 1) employs these higher separation altitudes

Finally, an alternative to the present concept of a central 200-400 volt solar array and individual electronic power processors for each thruster has been explored By dividing the array into eight sections, with each section configured to directly power an operating thruster, the following significant benefits are possible

1 A reduction in system failure rates
2 Increased subsystem efficiency from 0 64 to 0 72
3 A reduction in SEPS system cost (by approximately $2M per stage)
4 Reduced subsystem specific weight
5 Reduction in AST/SRT development requirements

In effect, stage performance is maximized while SEPS program costs are minimized

Powering an ion engine directly from a high-voltage solar array has been examined before and always discarded because of complex power and voltage switching necessary to cope with the wide range of heliocentric distances encountered on interplanetary missions However, with the increased emphasis on GEOSEPS operations, which all occur at 1 AU, the design requirements simplify

Directly powered electrical propulsion (DPEPS) in its simplest conceptual form is diagrammed in figure 6 2-14 It features the following assemblies

1 Eight solar-array subpanels, each configured to operate one thruster
2 Sequencing and switch gear (8) for startup and redundancy utilization
3 Vaporizer feedback control units (8) for thruster control
4 Conventional thrusters (10), thrust vector control, and propellant management

The solar-array stringing concept is illustrated in figure 6 2-15 for the power supplies identified in table 6 2-5 Solar arrays of the thruster-required voltage ranges have been built and tested However, operational compatibility with the ion engine has not been demonstrated at this time

We conclude that the DPEPS is a feasible concept for the Earth-orbital SEPS Significant improvements in performance, cost, and reliability are possible The solar-array technology is within reach and only relatively minor thruster redevelopment is anticipated Further study of this concept is recommended, and completion of array/thruster compatibility testing is strongly urged

Further details of the subsystem design investigations conducted during this study are given in Volume III of this final report

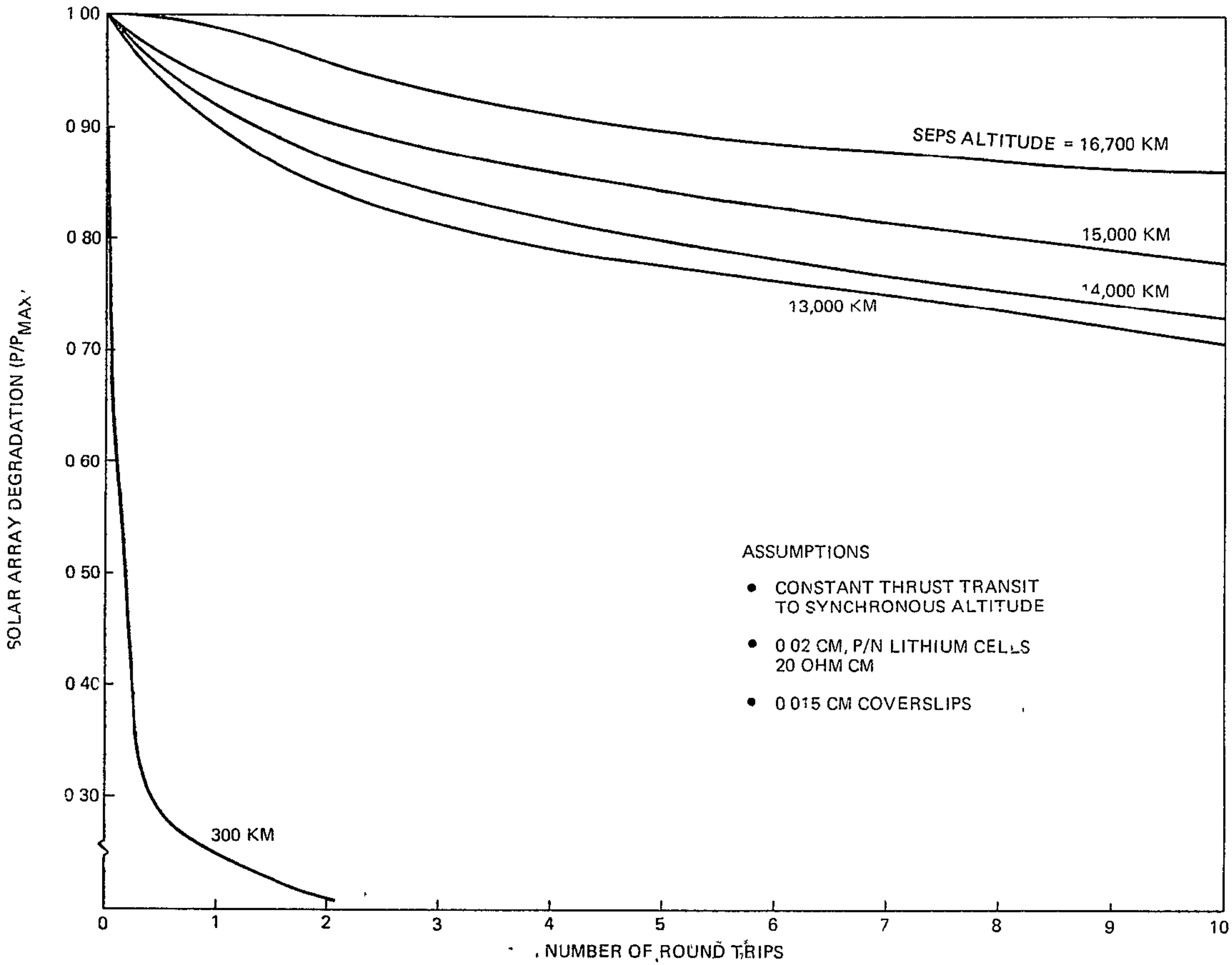


Figure 6 2-13 Solar Array Degradation on Geoseps

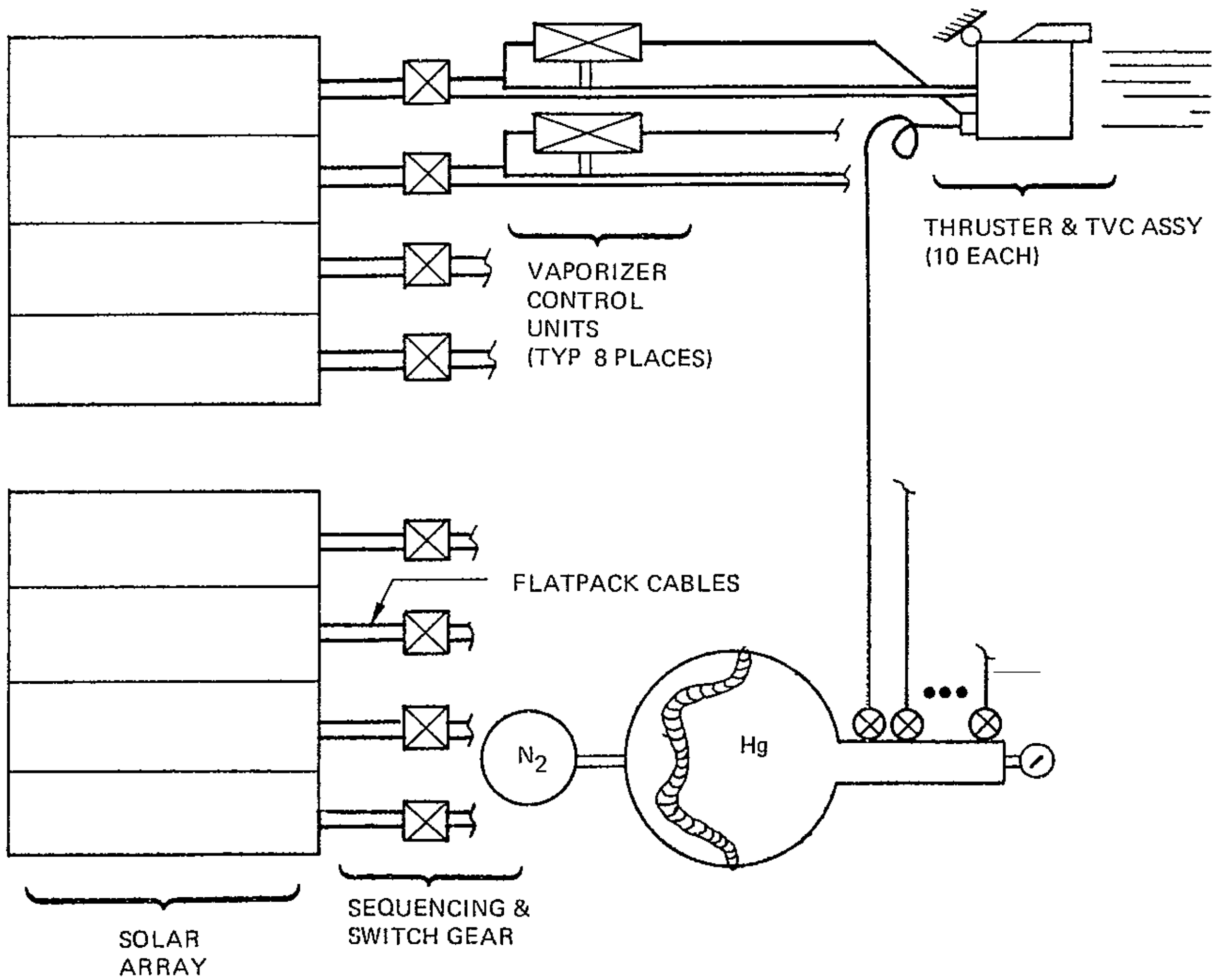
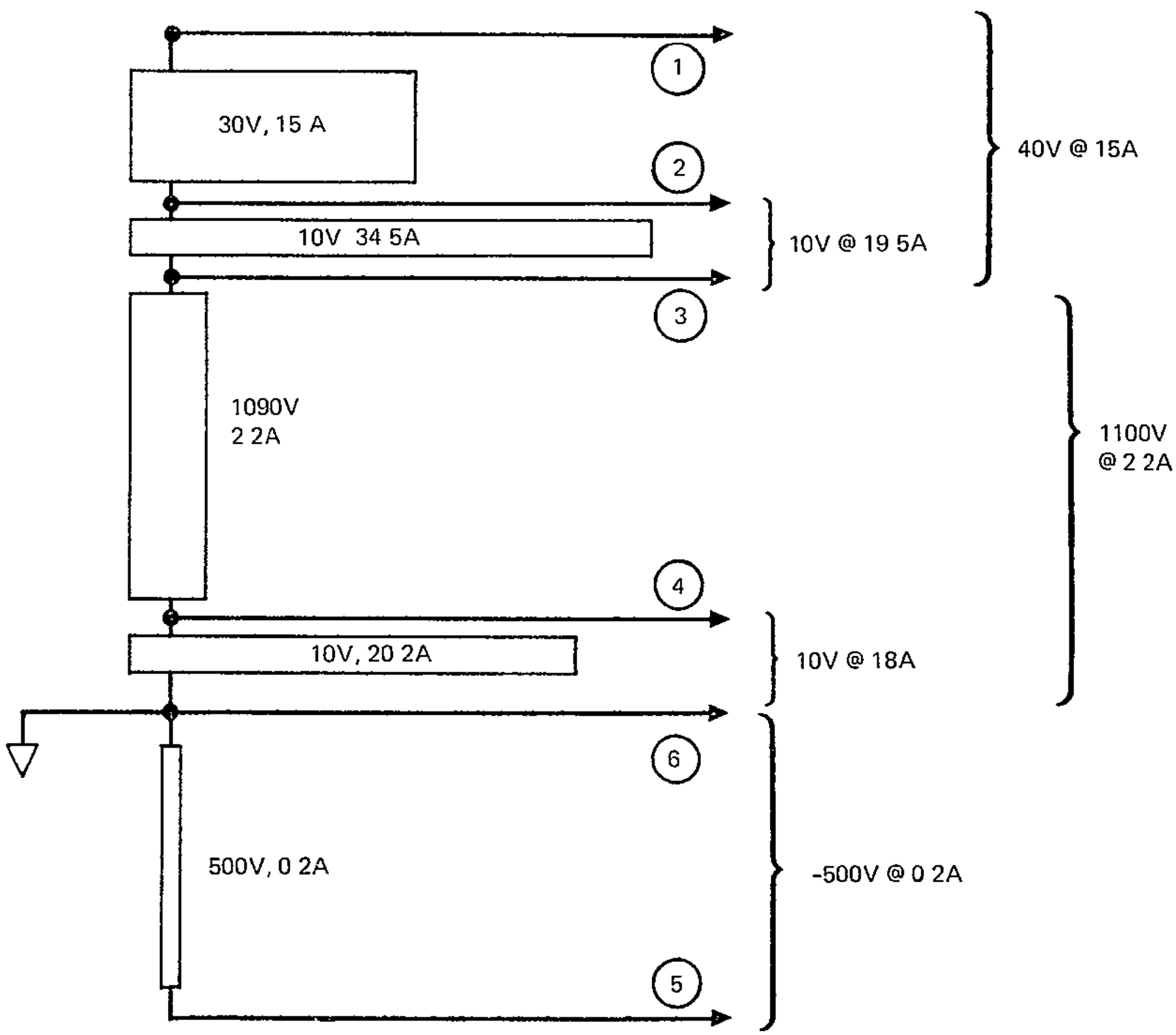


Figure 6 2-14 Directly Powered Ion Thruster Electric Propulsion System



NOTE A DEDICATED CIRCUIT IS PROVIDED FOR EACH THRUSTER

Figure 6 2-15 High Voltage Solar Array Circuit for 30-CM Ion Thruster

Table 6 2-5 30-CM Thruster Power Supply Requirements

SUPPLY NO	SUPPLY POTENTIAL (REF PPU GRD)	SUPPLY	MAXIMUM POWER (WATTS)	MAXIMUM	NOMINAL LEVEL
1	1100	ANODE	600	40V AT 15A	37V AT 14A
2	1100	MAIN ISOLATOR	20	10V AT 2A	4 5V AT 1A
	1100	CATHODE ISOLATOR	20	10V AT 2A	(8 2V AT 3 6A)
	1100	CATHODE HEATER	90	10V AT 9A	9V AT 4 5A
	1100	CATHODE KEEPER	60	10V AT 6A	10V AT 0 5
	1100	VAP FEED LINE	5	10V AT 0 5A	10V AT 0 5A
3	0	SCREEN	2420	1100V AT 2 2A	1100V AT 2A
4	0	MAIN VAPORIZER	20	10V AT 2A	7V AT 1A
	0	CATHODE VAPORIZER	20	10V AT 2A	3 5V AT 1A
	0	NEUTRALIZER VAPORIZER	20	10V AT 2A	3 5V AT 1A
	0	NEUTRALIZER KEEPER	60	10V AT 6A	10V AT 2 4A
	0	NEUTRALIZER HEATER	50	10V AT 5A	8 5V AT 4 2A
	0	MAGNETIC BAFFLE	10	10V AT 1A	10V AT 0 8A
5	0	ACCELERATOR	100	-500V AT 0 2A	500V AT 0 008A

7 0 IMPLICATIONS FOR RESEARCH

To ensure the timely development of the SEPS program, additional research should be considered The items identified below affect the requirements for the design, development, and operational phases of the program Accomplishment of these development programs would decrease the risks associated with a SEPS production program Table 7 0-1 provides a summary of these suggested items, including item title, reference paragraph, and estimated manpower and cost Each item discussed in this section is a summary of the more detailed discussion provided in section 4 0 of Volume IV

7 1 HIGH-VOLTAGE SOLAR ARRAY

High-voltage solar array studies and preliminary tests have demonstrated the practicality of developing a solar array tailored to the specific power requirements of the SEPS mercury thrusters This approach substantially simplifies the power processing unit and greatly increases overall system efficiency and reliability over the present-day power processor systems System cost would be reduced, resulting in a significantly more cost-effective system

7 2 DYNAMIC ANALYSIS OF LARGE SOLAR ARRAYS

To date, the largest flexible substrate solar array flown was designed to generate 1 5-kilowatt power Extensive dynamic analysis was performed to determine the interaction between that array and the flight vehicle The large SEPS array frequency will be an order of magnitude lower than the 1 5-kilowatt array Therefore, a dynamic model of the prestressed solar array is required to characterize the SEPS array dynamics and, to assess the flexible structure/control system interactions

7 3 ELECTRIC THRUSTER LIFE AND RELIABILITY IMPROVEMENT

Thruster life and random failure rate dictates the design of the stage propulsion configuration and the attendant probability of mission success Current technology indicates that a 20,000-hour life is readily attainable by upgrading the 10,000-hour-life thruster Random failure rate predictions for reliability calculations have been made by an analysis procedure with only minimal test data for support It is highly desirable to expand the current thruster development to include statistical lifetime and reliability testing

7 4 ELECTRIC THRUSTER STOCHASTIC PERFORMANCE CHARACTERISTICS DETERMNATION

Ion thruster performance uncertainties (steady bias and time-dependent variances) are virtually unknown These uncertainties dictate the amount of reserve propellant required to ensure mission success Current estimates of the required reserve range are from 5 to 15 percent, which could significantly affect SEPS performance and mission design Consequently, it is highly desirable to measure the variances and evaluate options for minimizing them

Table 7 0-1 Additional Research Items

TITLE	REFERENCE PARAGRAPH	PROGRAM MAN--YEARS	ESTIMATED PROGRAM COST ($1000)
HIGH VOLTAGE SOLAR ARRAY	7 1	5.5	492
DYNAMIC ANALYSIS OF LARGE SOLOR ARRAYS	7 2	7 0	688
ELECTRIC THRUSTER LIFE AND RELIABILITY IMPROVEMENT	7.3	8 0	427
*ELECTRIC THRUSTER STOCHASTIC PERFORMANCE CHARACTERISTICS DETERMINATION	7 4	1 75	90
EMI CHARACTERISTICS OF OPERATING THRUSTERS	7 5	1 5	97
MERCURY PLUME EFFECTS	7 6	2 25	135
MERCURY PROPELLANT SLOSH DAMPING TECHNOLOGY	7 7	3.0	170
EFFICIENCY AND RELIABILITY OF HEAT PIPE JOINTS	7 8	1 5	114
DIFFERENCED - DATA TRACKING TECHNIQUE DEVELOPMENT	7 9	3 0	142

*ASSUMES "ELECTRIC THRUSTER LIFE AND RELIABILITY IMPROVEMENT" IS A PARALLEL EFFORT

7 5 EMI CHARACTERISTICS OF OPERATING THRUSTERS

Measurements of the electric and magnetic field produced by present-day (series 400 and newer) electric thrusters have been limited to the static (nonoperating) condition Since payloads and SEPS subsystems must operate in the thruster generated fields, it is mandatory that the electromagnetic interference characteristics for an operating thruster be defined With this data, techniques such as shielding and suppression can be employed to ensure equipment operation

7 6 MERCURY PLUME EFFECTS

SEPS mercury plume ionization effects are anticipated to cause radio frequency (RF) signal attenuation and random modulation of the uplink and downlink RF carriers Accurate knowledge of the attentuation is essential in predicting the SEPS communication link performance Possible degradation to both the data channel performance and SEPS command receivers affects the ability to acquire and maintain track during the propulsion burn periods

Test data for communication links operating with mercury propellant systems do not exist, and analytical data are very limited A program has been defined to provide both the analytical and actual test data necessary to ensure satisfactory SEPS communication, command, and tracking links

7 7 MERCURY PROPELLANT SLOSH DAMPING TECHNOLOGY

Mercury propellant mass can be a large fraction of SEPS mass If it is sloshing in its tank, due to the high fluid density and the low accelerations present on a SEPS, the entire vehicle (which is characteristically very elastic because of the solar array) may respond with large oscillations This condition would adversely affect star tracking, steering, communications, and science operations

Technologies for damping mercury slosh energy have not been tested A test program to demonstrate slosh damping prevention technology for mercury storage tanks is recommended

7 8 EFFICIENCY AND RELIABILITY OF HEAT PIPE JOINTS

Heat pipe cold plates and variable conductance heat pipe radiators have been examined through flight experiments as potential solutions for spacecraft temperature control of high power density electronic modules They offer increased temperature control capability along with long life, high reliability, and low weight in comparison to current design practices utilizing louvered radiators In addition, their interfaces can provide a passive thermal management capability between electronic cold plates

To ensure proper development of this approach, a program for a multiple heat pipe temperature control and thermal energy management subsystem has been outlined

7 9 DIFFERENCED-DATA TRACKING TECHNIQUE DEVELOPMENT

SEP thruster process noise is anticipated to be a major navigation error source for all missions and a success-critical factor on planetary missions with high delivery accuracy requirements Differenced-data tracking techniques such as Quasi Very Long Baseline Interferometry (QVLBI) doppler and Simultaneous Interferometry Tracking Technique (SITT) offer the potential for dramatic reductions in process noise error effects Development of these techniques to date has been limited to design of the required straightforward software and partial operational feasibility tests on Mariners 9 and 10

A development program, including firm commitments for test on the Viking-Orbiter '75 and the Mariner Jupiter/Saturn '77 missions, is required to ensure availability of this tracking capability in the early 1980's time period

8 0 SUGGESTED ADDITIONAL EFFORT

Primary follow-on effort should address three basic areas First, the technical feasibility of using SEPS for geosynchronous missions should be further substantiated Second, development of a payload interface definition to allow assessment of the impact between SEPS and its payloads should be investigated Third, techniques that would potentially reduce the SEPS production cost should be evaluated

8 1 GEOSYNCHRONOUS REQUIREMENTS VERIFICATION

After review of the Earth-Orbital Traffic Model for SEPS, which was developed from the January 1974 revision to NASA TMX 64751, "Space Shuttle Traffic Model," it was determined that the SEPS configuration developed for planetary missions could perform the Earth-orbital missions However, SEPS size, configuration, and economic parameters should be reviewed, assuming the baseline mission is an Earth-orbital mission in lieu of a planetary mission The more important subjects within these categories are discussed briefly in this section

8 1 1 Earth-Orbital SEPS Size

The optimum size SEPS should be determined for Earth-orbital applications Radiation damage and cell efficiency loss considerations for the various solar cell types may indicate that the current 25-kilowatt solar array is not the optimum for this mission Along this line, high-voltage solar arrays should be examined, since they eliminate the need for the power processors and the attendant efficiency loss The optimum number of operating thrusters should be determined and the solar array sized to provide the necessary power for the stage life In this effort, payload size and delivery time constraints should be considered If delivery timing is not critical, then additional delivery time allows an acceptable workaround when thrusters fail Initial evaluation of the payload traffic model indicates that this option is available and should be thoroughly analyzed in future efforts, to take advantage of its effective increase in redundancy

8 1 2 Payload Changeover Orbit

The payload traffic model should be examined to determine the optimum changeover altitude Tug/SEPS performance and timelines should be considered for a sortie with five payloads up to geosynchronous orbit and return of four spent payloads from that orbit With these data and solar array degradation considerations, the optimum tug size can be determined that would provide a changeover orbit altitude consistent with attainment of the desired payload trip time This model should be compared to an optimum traffic model using only an interim upper stage (IUS)

8 1 3 Thrusters Considerations

Reliability considerations for thrusters require additional test data to optimize the number of thrusters in the SEPS configuration Only limited random failure rate and usefull life data exist, and these are key areas for meaningful evaluation and success predictions Also, requirements must be established that define the number of thruster failures allowed before the mission/sortie is considered unsuccessful Thrusters could fail, but the payloads could still be delivered at some increased time interval Payload delivery time constraints that would limit the number of thruster failures allowed should be identified

8 1 4 Docking

SEPS must dock with the Tug initially for transportation to changeover orbit and for refueling and payload transfer during follow-on sorties It must dock with multiple payloads for delivery to geosynchronous orbit and retrieval from that orbit The Tug must dock with SEPS and with payloads for delivery and retrieval from the changeover orbit SEPS/Tug operations interface and mechanisms that will perform the required functions during these dockings require further definition Use of manipulator arms to aid the payload exchange function should be compared to complete docking frames on each payload The low structure moment of inertia inherent in manipulator arms, which leads to dynamic instability, must be traded against the weight and space penalty required for individual docking frames for each payload

Different docking techniques should be analyzed, including the cost of each approach, time period required for the complete docking and exchange function, changing center of gravity effects for each docking method, and maneuver methods available for the required docking procedure

8 1 5 Refurbishment

The Boeing SEPS design approach is to design a highly reliable stage that is not likely to experience a failure that renders it inoperable for its intended use With this approach and study time limitations, only a cursory SEPS refurbishment analysis was performed More detailed consideration is required using the optimum payload traffic model and refined cost, reliability failure rate, and useful life data Subsystem component replacement schedules should be redeveloped for the life of the stage This requires thorough examination of each mission phase and development of probability of mission success versus cost for refurbishment data Operational versatility must be examined to determine when to replace and recover the stage for refurbishment Certain equipment failures could be tolerated by revising the traffic model, thus delaying or possibly eliminating the need for refurbishment Thrusters and power processors are examples of equipments in this category Cost estimates for each phase of the refurbishment cycle should be developed in greater detail This data should all be considered in determining at what point and at what level refurbishment would be cost effective

8 1 6 Shuttle Installation

Shuttle installation of the Tug/SEPS/Payloads and Tug/Payloads alone must be further defined Mounting and handling of these combinations must be considered for delivery to Shuttle altitude and return of spent systems from that altitude Load path effects for independent unit mounting versus cantilever stack should be examined Provisions must be made for returning mercury contaminated equipment to the ground To avoid a possible crash hazard, an ability to dump the remaining mercury and hydrazine is required

8 1 7 Stage On-Orbit Refueling

Mercury and hydrazine on-orbit refueling can be accomplished by use of a fluid transfer technique or through actual tank interchange Various methods within each approach should be further traded to aid selection of the most cost-effective approach for this function Complexity, weight, volume, reliability, cost, and operational utility should be fully considered for each method

8 2 PAYLOAD INTERFACE

SEPS design must interface with numerous different payloads Payloads must be grasped, serviced, coddled and/or set into motion The attendant procedures must necessarily set requirements on both the SEPS and the payload systems

8 2 1 SEPS Payload Support Requirements

Payload support requirements must be developed for individual and multiple payloads These requirements must be negotiated between the SEPS designers and the payload designers

An attempt should be made to minimize constraints levied on payloads while limiting the penalty to SEPS Support items considered should include power, commands, fluids, and telemetry requirements Spin up prior to release and separation rate and attitude requirements must be developed This effort should also consider the applicability and effectiveness of the SEPS providing servicing on the way up to geosynchronous orbit and return Another possible support application for consideration would be to use SEPS as a power source for semi-failed spacecraft

8.2 2 SEPS/Payload Physical Interface

SEPS/Payload physical interface requires further definition The methods selected for payload deployment and retrievel will greatly affect this effort If manipulators are used the docking adaptors could be smaller and less complex However, manipulator arms are inherently low structure moment of inertia, which leads to dynamic instability Other procedures for the exchange function, such as end-to-end docking, require more elaborate docking adaptors Weight and space penalties are associated with this approach Docking aids for the various docking methods must be considered For instance, a manipulator system would probably require more aids than the end-to-end docking approach Other physical considerations should include the necessity of umbilicals, their location and size Servicing and environment related physical interface requirements should be identified

8 2 3 Payload Characteristics

Payload characteristics are important in evaluating, handling, stacking, and delivery procedures They are the prime input for assessing the feasibility and practicality of multiple up/down payload combinations To date, payload characteristics are defined in the document "Summarized NASA Payload Descriptions, Automated Payloads," July 1974 This document provides general information on payload weight, length, maximum diameter, and power More descriptive information and additional parameters should be developed The diameter of both ends of the payload should be provided, if it differs Center of gravity and stability requirements are important Structural descriptions and the ability to mount docking adaptors to both ends is important data Possible constraints such as docking loads and dynamics, trip time, and those resulting from sensor servicing and environment-related requirements should be identified

8 3 SEPS POTENTIAL PRODUCTION COST REDUCTION TECHNIQUES

It is important to continually search for cost reduction techniques For SEPS, a number of potential production cost reduction techniques require further consideration Solar-array production methods, use of high voltage solar arrays to eliminate the need for the power processors, increased use of the onboard computer to eliminate or at least greatly reduce the need for ground support equipment, and development of a more optimum production schedule are examples of areas that warrant further evaluation

8 3 1 Solar-Array Production Techniques

Current cost estimates indicate that the solar array is by far the most costly component of the stage Possible methods for reducing this cost should be fully investigated For example, the ability to use larger hexagonal solar cells could significantly reduce production cost Developing an automatic method for installing solar cells on the substrate would reduce labor cost Another material and labor saving area would be to develop a simple spray cover or glue-down plastic cover for the cells

8 3 2 High-Voltage Solar Arrays

High-voltage solar arrays with integral power conditioning would provide regulated d c power directly to the ion thrusters This eliminates the need for the current power processor units, resulting in greater power system efficiency, increased system reliability, and reduced system weight and cost A program has been outlined to develop this approach A summary of this is provided in paragraph 7 1 of this volume and a detailed discussion is provided in Volume IV, section 4 0

8 3 3 Utilize On-Board Computer for System Tests

The current test approach requires accessible test connections and ground support equipment and facilities for vehicle system/subsystem testing It may be possible to add more remote multiplexers to the stage data system and eliminate the need for these directly accessible test connections This potential approach dramatically reduces the requirement for ground support equipment and facilities Performing the test with the stage data system also reduces test and cost by providing a more automatic vehicle checkout An attendant feature of this approach is that the capability for on-orbit malfunction diagnosis/work-around would be greatly enhanced

8.3.4 Optimize Production Schedule

Current program ground rules do not optimize the production schedule for least cost A period of 52 months exist from the first funding in late 1976 to first launch in March, 1981 On the Mariner 10 program, a like effort was accomplished in 30 months Assuming that solar array and propulsion technology is continuing and will be complete by Phase C/D award, a more cost effective condensed period should be investigated Also a 3-year pause between production demand groupings creates a costly situation Holding personnel through the 3-year period or retraining for the later phase are costly approaches Spreading production throughout the 3-year period results in storage periods up to 2-1/2 years Additionally, this latter method requires three stages in early 1981 and then dramatically slows to one stage every 9 months Revisions to the program or mission schedule should be considered to eliminate this costly approach

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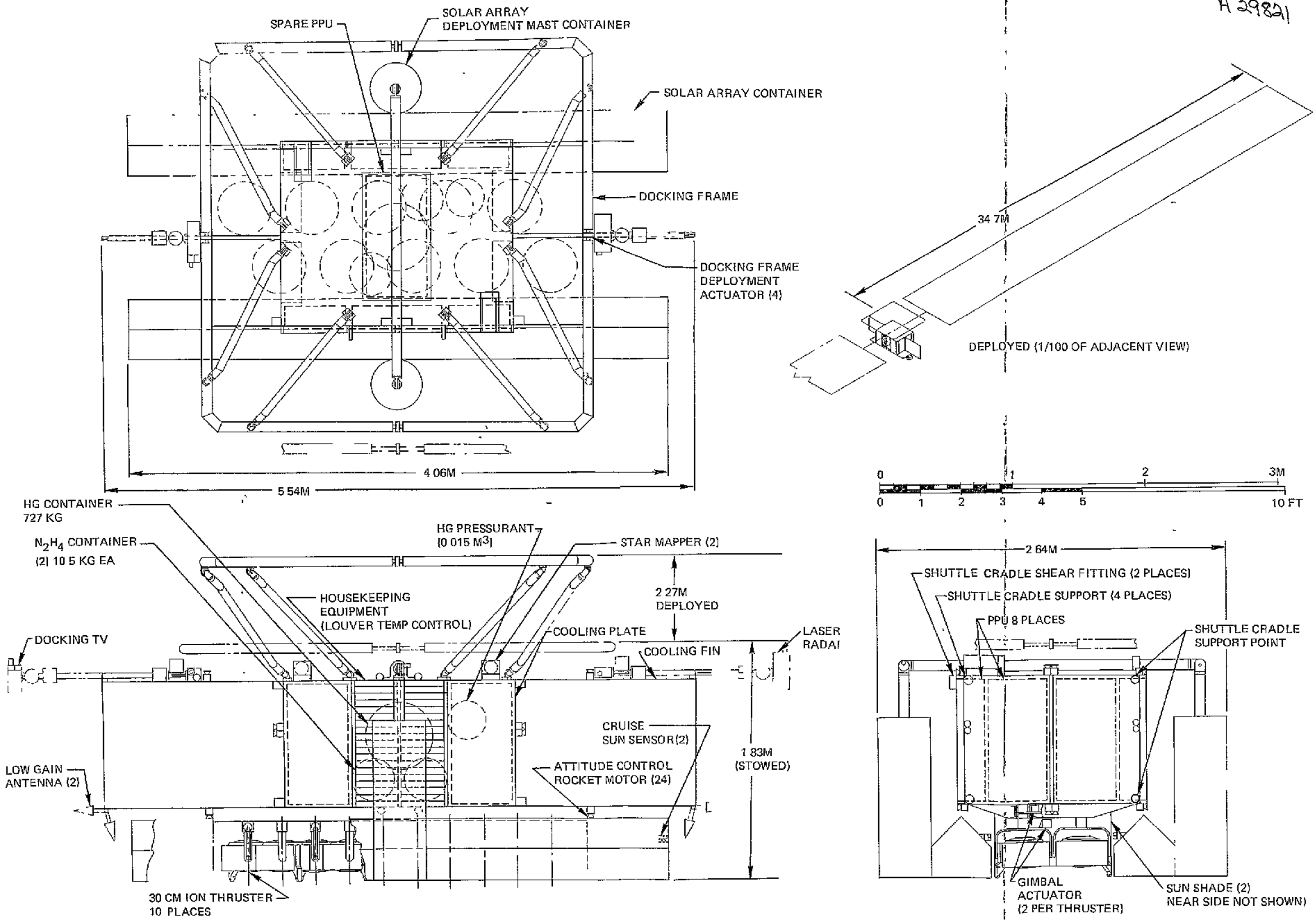

Figure 6 0-1 General Arrangement – SEPS Model 1038-7/EO

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66b

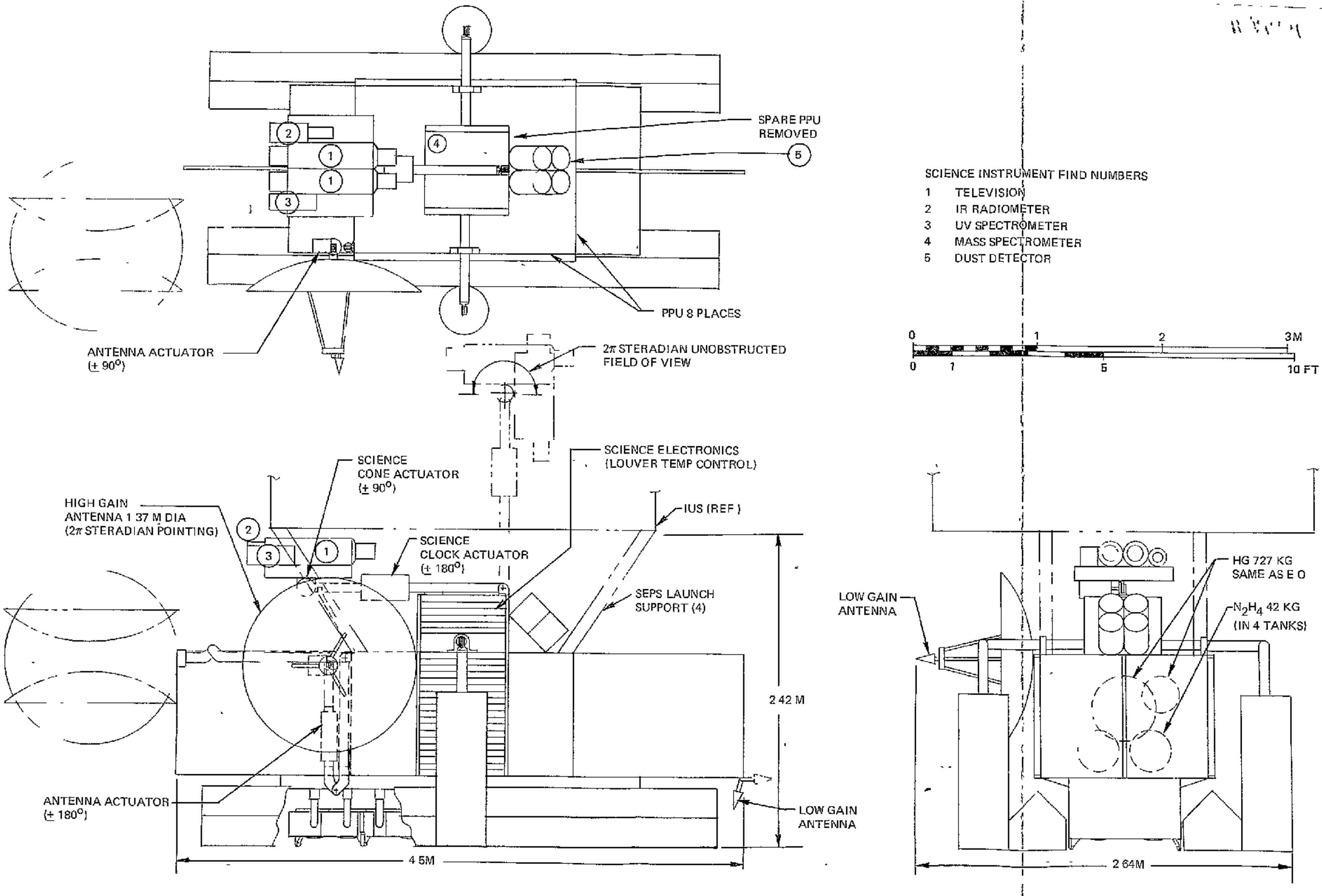

Figure 6 0-2 General Arrangement — SEPS Model 1038-7/PLAN

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67b

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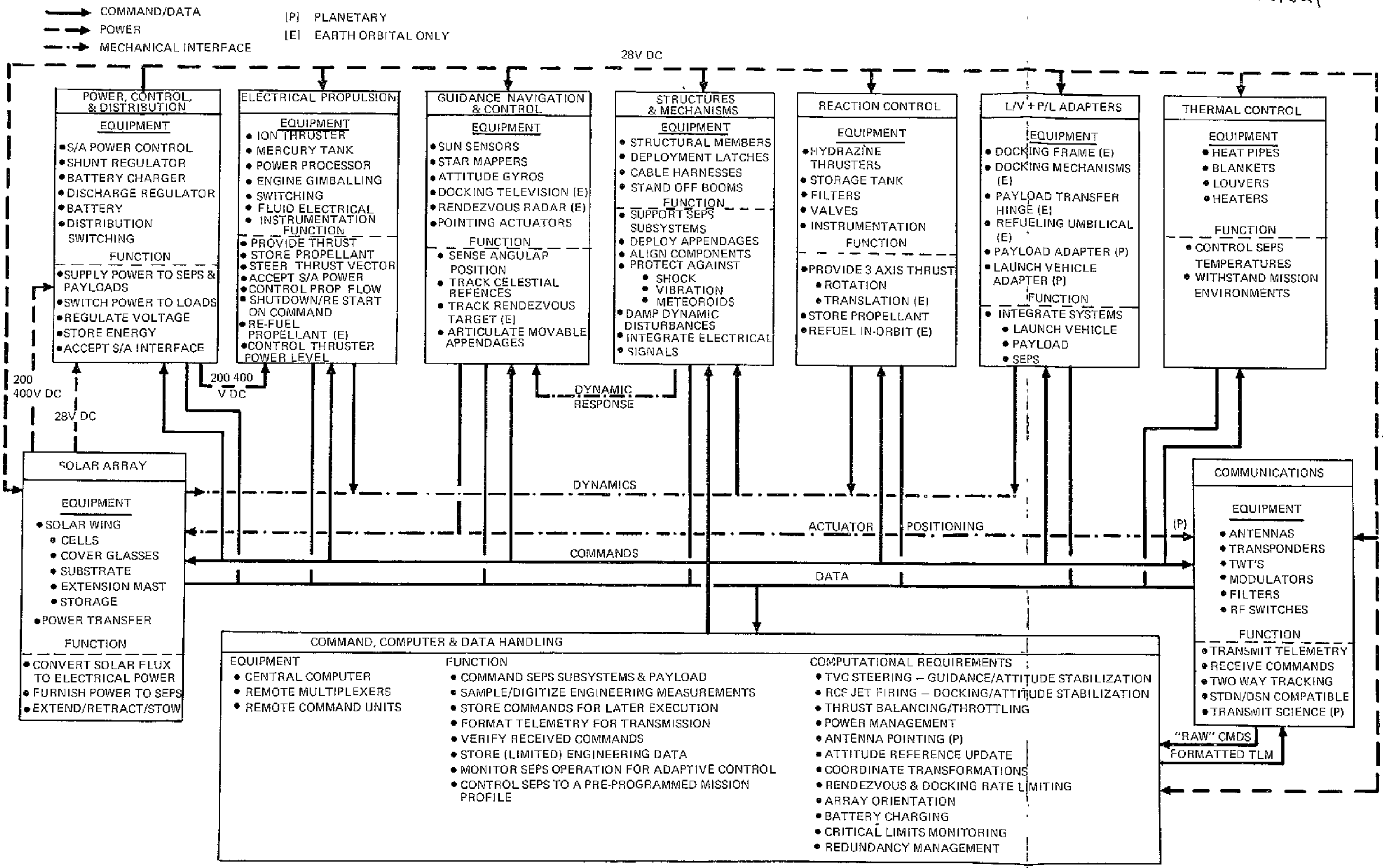

Figure 6 1-7 System Functional Block Diagram

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68 b

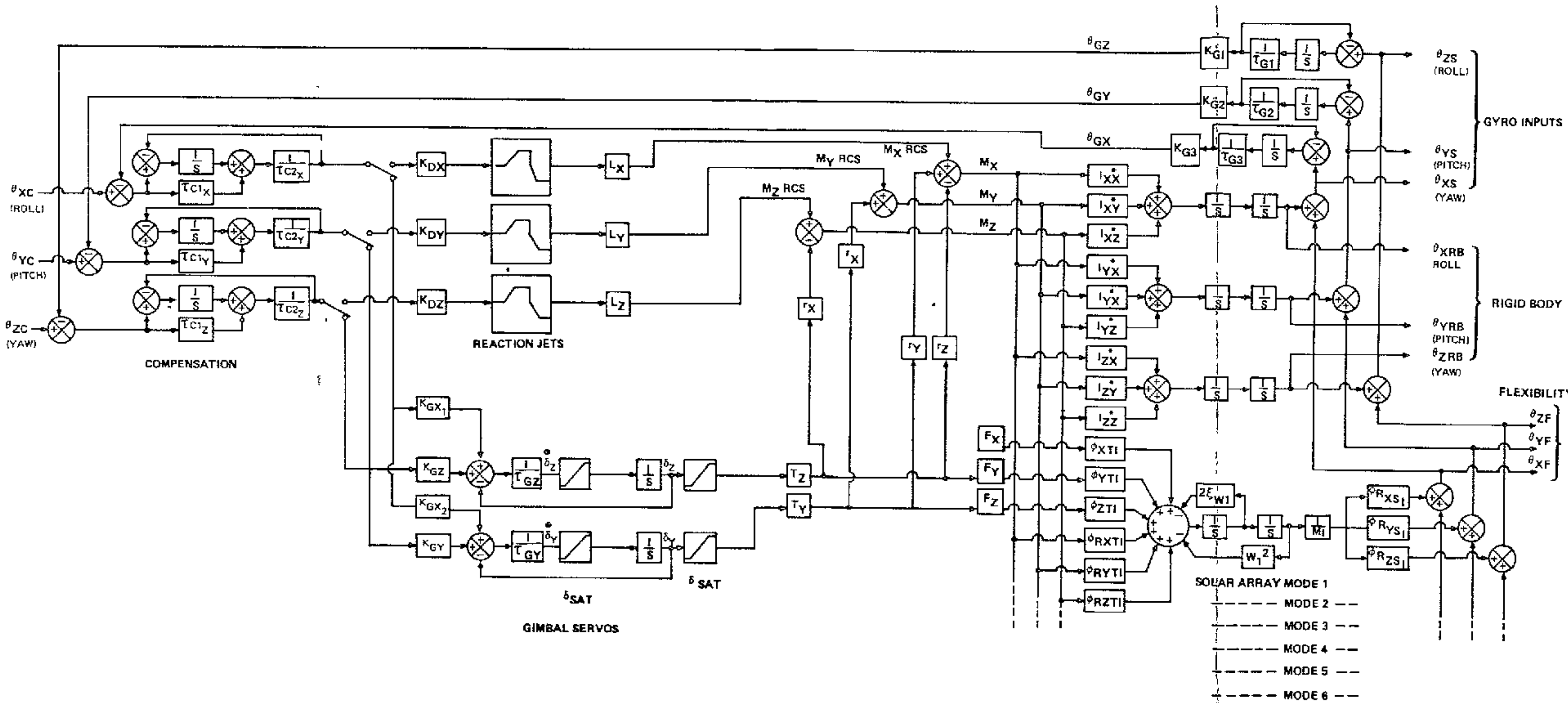

Figure 6 2-2 SEPS Stability Model

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